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AN EVALUATION OF FATIGUE LIFE IMPROVEMENT PROCESSES

BRUCE C. GALT

Structural Engineering and Design Co.
Los Angeles, California

TECHNICAL REPORT AFFDL-TR-68-138

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AN EVALUATION OF FATIGUE LIFE IMPROVEMENT PROCESSES

BRUCE C. GALT

FOREWORD

This report on the A-26A fatigue life improvement processes was prepared by the Structural Engineering and Design Co., a Division of On Mark Engineering Co., Los Angeles, California, under the supervision of Mr. Bernard Kreitzer. The program was initiated under Project No. 943 D for the A-26A aircraft by the Special Projects Branch, ASZSC, of the Aeronautical Systems Division, Air Force Systems Command, Wright-Patterson Air Force Base, Ohio, and conducted under Air Force Contract No. F33657-68-C-0112. Mr. Howard Wood of the Air Force Flight Dynamics Laboratory, FDTR, was the Air Force technical monitor. This report covers work conducted from 5 September, 1967 through 3 September, 1968. This report was submitted by the author on 4 October, 1968.

The author gratefully acknowledges the assistance given by Mr. Leo E. Klar, Mr. Richard J. Rose, and Miss Muriel C. Smith.

In addition, the author wishes to acknowledge the contributions made to this program by Mr. Berton Bland of Materials Testing Laboratory, Magnaflux Corporation, Los Angeles; Mr. F.E.Van Vlack, Vanaero Co., Los Angeles; Mr. Sanford Friezner, Specialized Testing Co., Los Angeles; and Mr. Paul Bickel, Metal Improvement Co., Los Angeles.

This report has been reviewed and is approved.

Francis J. Janik Jr.

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Chief, Theoretical Mechanics Branch
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ABSTRACT

This report presents the results and conclusions of a specimen testing program established to confirm or modify certain conclusions reached during the cyclic test of the A-26A wing and which affect the A-26A Airplane Service Fatigue Life Prediction.

The object of the program was to evaluate the effects of reaming existing fatigue-critical bolt holes to larger diameters and peening the metal surfaces inside of and adjacent to the enlarged holes.

Specimens were designed to duplicate the conditions of the fatigue-critical portions of the A-26A wing. A series of tests were run, changes were made in the program schedule as the result of information gained, and a final series of tests were conducted.

It was concluded that (1) the damage reduction due to the reaming process produced results very nearly as originally considered in the A-26A Service Life Prediction, and (2) the reduction in damage accumulation rates of the A-26A fatigue test wing, originally attributed to the effects of peening, was actually caused by an increase in bolt preload achieved upon installing larger diameter bolts after the reaming process.

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SYMBOLS

C	dimensionless bolt torque factor
D	hole diameter
D_o	damage accumulation at the edge of a hole prior to reaming the hole to a larger diameter
ΔD	reduction due to reaming of the magnitude of the damage accumulation at the edge of a hole
f_{br}	applied bearing stress
f_{net}	applied tension stress based on net area
f_{max}	calculated maximum tension stress at a point on the edge of a hole
F_{tu}	allowable ultimate tension stress
F_{ty}	allowable yield tension stress
e	ultimate elongation
E	distance from center of hole to edge of part
K_t	stress concentration factor
n	number of applications of a given loading cycle
N	number of loading cycles to failure
P	applied load
s	sample standard deviation - a measure of scatter in statistical data
S	maximum applied cyclic stress
$\overline{S}$	endurance limit stress
t	thickness
TS	tensile strength - a constant used in S-N curve derivations
W	width

NOMENCLATURE

basic life - log average of test cycles to failure for a given specimen design tested at a constant loading without any processes applied

cycle ratio - n/N - ratio of the number of applications of a given loading cycle to the number of applications of that loading cycle which produces failure.

linear cumulative damage rule - the assumption that the summation of cycle ratios is equal to 1.00 at failure

damage accumulation - existing fatigue damage at a given location on a structure due to the application of cyclic stresses. (Explicit evaluation of this quantity is not possible by either analytical or experimental techniques. Average test results or cumulative damages theories are employed to approximate values used for analysis of given structural members. Damage quantity is expressed as a fraction of failure and is equal to 1.00 at initial crack.)

damage reduction - ΔD - reduction due to reaming of the magnitude of the damage accumulation at the edge of a hole

damage reduction factor - $\Delta D/D_0$ - ratio of damage reduction due to reaming to the damage accumulation existing at the edge of the hole prior to reaming

life ratio - $\Sigma n/N$, with all applied cycles, n, included regardless of processes applied after the start of cycling, and with N equal to the basic life (without processes)

.03 (.06) inch ream - the process of increasing the diameter of a hole, measured as the difference between the new diameter and the old diameter

zero-time - to remove all existing damage by reaming

probability plot - a plot of log N versus per cent failure using a probability grid (probability paper) which produces a straight line for a normal (Gaussian) distribution of data

Almen intensity - degree of peening intensity determined by peening one side of a metal strip and measuring the curvature produced by the peening

Open Hole, Loaded Hole, and 4 (8) Bolt Joint - specific specimen designs

NOMENCLATURE

(Continued)

Interim Repair, Permanent Repair - specific modifications installed on A-26A service aircraft to increase the fatigue life capability of the wing structure

Service Life Prediction - final report (Reference (1)) of the A-26A cyclic test program, giving a predicted damage accumulation for critical stations of the wing due to a given airplane mission utilization

SECTION I

INTRODUCTION

Generally when a fatigue problem occurs on an operational airplane, the troublesome area is reworked and a repair is designed and installed. The repair procedure usually consists of two basic requirements. First, a removal of cracked or damaged material in the region of the failure, and secondly, the addition of parts to reinforce the area and reduce the magnitude of the cyclic stresses causing the damage.

Although the overall design of a repair of this type is within the state-of-the-art of competent aircraft structural designers, little has been done to evaluate the effects of the individual processes used in such a repair.

An evaluation of the effectiveness of individual repair processes is necessary in order to provide a basis for a Using Command to determine whether to effect a minor repair process and return the airplane to service, or whether to restrict the airplane and ask for a modification design.

The general objective of this testing program is to determine the effectiveness of two basic repair processes for the purpose of extending the fatigue limitations of airplanes in service. These processes are (1) the removal of fatigue damaged material around small bolt holes by reaming to a larger diameter, and (2) glass peening in and around small attachment holes.

The specific objectives of this testing program are, given an aluminum part with critical small screw holes:

(a) To determine the effect upon fatigue life of reaming and/or peening after some initial damage has been accumulated.

(b) To determine the most advantageous sequence of events for extending fatigue life.

(c) To determine what changes (if any) should be made to the Life Prediction of the A-26A aircraft as a result of this testing program.

In the process of meeting the above objectives, it was decided to add a fourth objective:

(d) To determine the effect of bolt preload, or tightening torque, upon the fatigue life of parts containing bolts loaded in shear.

The objectives for this testing program were formulated as a

result of the A-26A wing cyclic fatigue test. During the course of that program, the skins were stripped from the critical areas of the lower spar caps, and the attachment holes in the caps were reamed to a larger diameter and shot peened. The structure was reassembled and the fatigue program life objective was achieved, and exceeded, by testing. The processes applied to the test wings were then applied to the fleet aircraft.

The effects of these processes were determined and used in deriving predicted fatigue damage accumulations for service airplanes. This analysis is included in the A-26A Service Life Prediction Report, Reference (1).

The purpose of the specimen testing program was to confirm or modify these evaluations as they apply to the A-26A Service Life Prediction, and also to present the results in general terms for utilization in evaluating problems which may occur on other aircraft models.

A previous investigation into increasing the fatigue life of existing structures was done by Butler, Reference (9). This report includes an evaluation of increasing rivet sizes in a sheet metal joint and the effects of removing cracked material in heavier structural members.

Reference (11) presents a summary of the uses and some of the improvements which can be gained by peening. Reference (2) contains test results for round specimens, peened and not peened, with bending and axial loads applied. No quantitative data has been found which applies directly to the effect of peening aluminum spar caps.

Reference (10) contains the only data found on the relationship between bolt preload and fatigue life. Page 54 of that report shows the increase in fatigue life with bolt torque for a thin sheet with high bearing stresses.

SECTION II

SUMMARY

1. Test Results

Sketches of the four specimen designs used in this program are shown in Figures 1, 2, and 3. Three basic designs were used: Open Hole, Loaded Hole, and 4 Bolt Joint. An 8 Bolt Joint, similar to the 4 Bolt Joint, was added after the start of the testing program.

The results of 791 specimen tests are recorded in Appendix II. Cycles to initial crack are recorded for 86 of these tests. Cycles to initial crack were recorded by use of a crack wire circuit which shut down the testing machine upon failure of a wire located 1/16 inch from the edge of the critical hole in the specimen. (Table II)

Test results were consistent in most cases, with the joint specimens producing the greatest variation in test results, as would normally be expected.

Test failures occurring at points outside of the critical areas were few, and failure cycles recorded only if the number of cycles were higher than the average for the group.

Appendix I includes summaries of basic S-N data. A comparison of basic S-N curves for the four specimen designs is shown in Figure 25 for a mean stress of 20,000 lb/in^2. Figure 26 gives the comparable curves for a mean stress of 10,000 lb/in^2.

Individual data points for various Open Hole specimen tests are shown in Figure 27. Figures 28 and 29 show data points comparing the two reaming processes, peening process, and basic data for Loaded Hole specimens and for 4 Bolt Joint specimens.

Appendix II contains one data sheet for each group of 16 specimens tested. S-N curves are compiled in Appendix III, one for each test except for those which had a requirement for constant load and variable bolt preload.

Unless otherwise noted, all curves in this report are based on log mean cycles to failure.

2. Reaming Process

Two reaming processes were investigated. The first consisted of increasing a screw hole by .03 inch diameter and installing a special 1/32 inch oversize diameter bolt, which had the same thread and nut as the smaller standard diameter bolt. The second process was the reaming of a hole to the diameter of the next larger nominal attachment size, an .06 inch increase in diameter, and installing a standard fastener.

The .06 inch increase in hole size produced a very favorable effect upon fatigue life. Not only was this ream more effective than the smaller ream, but in addition, the installation of the larger bolt with the normal increase in standard preload caused the fatigue life to increase significantly as shown in the chart of Figure 12.

Although the .03 inch increase in hole diameter was effective in many cases, it did not prove to be completely reliable, particularly with the high bearing stresses encountered in the joint designs. Furthermore, the .06 inch increase produced results which indicate the obvious advantage of this process as compared to the .03 inch ream.

3. Peening

No increase in fatigue life could be attributed to the application of the peening process. In fact, in cases where fatigue life was very sensitive to bolt preload, peening was shown to be detrimental (Figures 15 and 17). In cases where sufficiently high bolt torque was applied, the fatigue life was brought up to the values achieved in tests with not peened surfaces (Figure 19).

The fatigue life improvement attributed to peening in the A-26A Service Life Prediction, Reference (1), has now been shown to be due to the effects of an increase in bolt preload as a result of installing larger fasteners after the reaming process.

4. Optimum Process

Analysis of the results of this testing program indicates that the best process for extending the fatigue life of a critical structural area containing small bolts or screws is as follows:

(a) Ascertain that the existing damage accumulation is not greater than 60 per cent of fatigue failure.

(b) Ream existing critical holes to increase the diameter by .06 inch.

(c) Install bolts or screws 1/16 inch larger in diameter than the original sizes.

(d) Apply torque values to the nuts of the new fasteners which are not less than industry standard values for shear applications.

If the existing fatigue damage accumulation is greater than 60 per cent of the predicted failure time, additional care should be exercised during the reaming process. Beyond 60 per cent damage accumulation, the probability of a crack existing, or of a condition where a crack is about to occur, increases rapidly.

A crack detection inspection should be made, and possibly an extra increase in standard bolt size should be considered. A more exact definition of applying a process to an area where damage accumulation is near 100 per cent is beyond the scope of this program and should be the object of additional investigations.

Damage reduction factors due to reaming are plotted in Figure 13, and Section III, 7, contains a discussion of the results produced by the reaming process.

5. Bolt Preload

The effects of bolt preload as produced by nut tightening torque were evaluated only as a necessary step in the process of determining the effect of reaming holes upon fatigue life. The plots in Figure 19 are therefore quite limited in scope. Further definition of these curves and the addition of other materials and ratios of bearing to tensile stresses should be the object of additional investigations.

It should be noted that bolt preload has been found to produce the most powerful effect upon fatigue life of any of the variables investigated in this program.

Figures 30 and 31 show a failure which is typical of several which occurred in the 4 Bolt Joint specimens. The effect of installing a 5/16 inch diameter bolt with normal preload caused the initial fatigue crack to start away from the minimum net section across the hole.

6. A-26A Life Prediction

The recommended modification to the A-26A Service Life Prediction damage accumulation is shown in Figure 24. This modification is the result of applying a new damage reduction factor to the .06 inch reaming process. This change produces a modification to the damage accumulation rate at the front spar for the last phase of the cyclic test. This adjustment is possible because it is now concluded that the damage rate change is a function of the combination of attachment size and attachment load, and not due to the effects of peening.

The modification to the Service Life Prediction is small (Figure 24) and will not significantly reduce the effectiveness of the A-26A fleet.

SECTION III

DISCUSSION

1. Specimen Design and Fabrication

The fatigue test specimen design is a simulation of the A-26A spar cap skin attachment flange. All of the critical fatigue failures which occurred during the A-26A wing cyclic test started at a spar cap flange in the inboard area of the wing. (Reference (1))

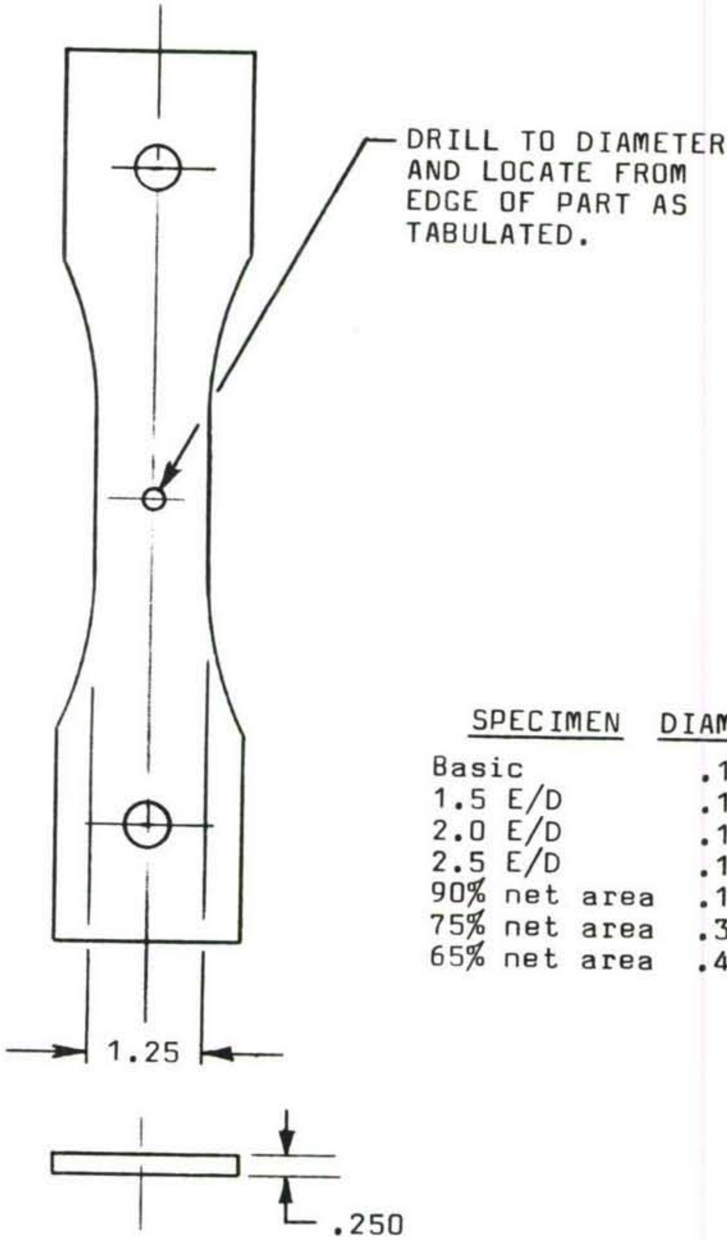
The flanges in this area are approximately .250 inch thick and 1.25 inch wide and the material is 2014-T6 aluminum alloy.

Three basic specimens were designed: Open Hole, Loaded Hole, and Joint. The Open Hole specimens were designed to test the fatigue life improvement processes without the complication of bolts and plates. Loaded Hole specimens simulated the effect of shear flow being transferred from a skin panel to a spar cap, with light to medium bearing stresses, while the Joint specimens provided relatively high bearing stresses such as would occur at a splice or at the end of a load carrying member, such as a heavy skin panel or reinforcement strap.

Open Hole specimens, Figure 1, include variations to account for the effects of net tension area, produced by the drilling of different hole diameters in a constant specimen width; and hole edge distance, by locating the hole off the center line of the specimen. Thickness variation was accomplished by fabricating a set of specimens from .125 inch sheet material as compared to the basic .250 inch material.

The Loaded Hole specimen, Figure 2, consists of a continuous specimen member with two steel doubler plates bolted to the surfaces, one bolt loaded in double shear at each end of the plates. The steel plates take load when the specimen is loaded due to consistent deflections of the plates and the specimen between the bolt locations. A light bearing load is applied to the specimen holes as the load applied to the plates is transferred from the specimen by the bolts.

The Joint design, Figure 3, consists of a specimen sawed across the center to form two separate pieces which are spliced together with two .125 inch thick aluminum splice plates. A total of four bolts are required, two on each side of the saw cut. The total specimen load must be transferred to the splice plates by two bolts, each loaded in double shear.



SPECIMEN	DIAMETER	LOCATION
Basic	.187	.625
1.5 E/D	.187	.280
2.0 E/D	.187	.374
2.5 E/D	.187	.467
90% net area	.125	.625
75% net area	.312	.625
65% net area	.437	.625

Figure 1. Open Hole Specimen Design.

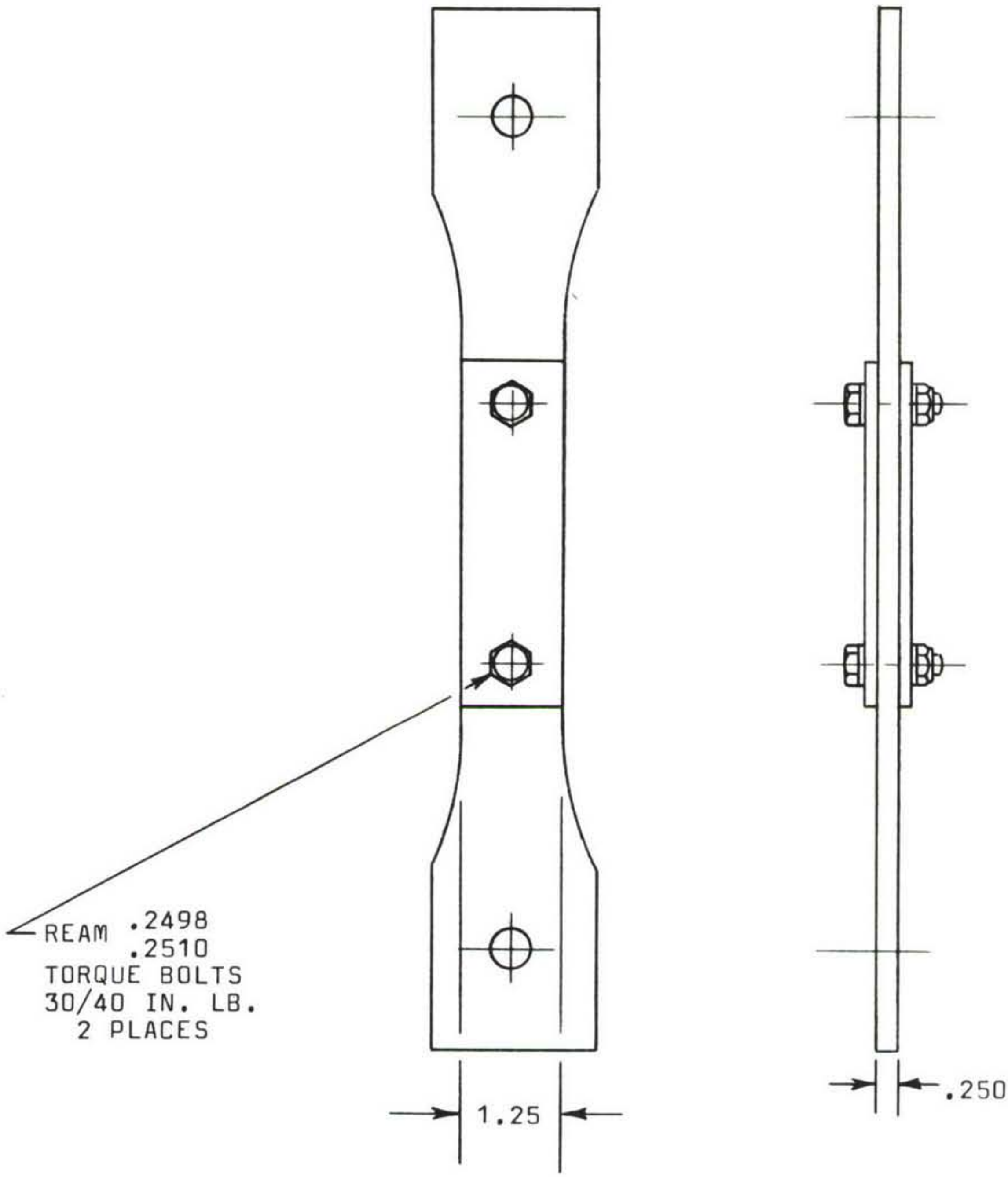


Figure 2. Loaded Hole Specimen Design.

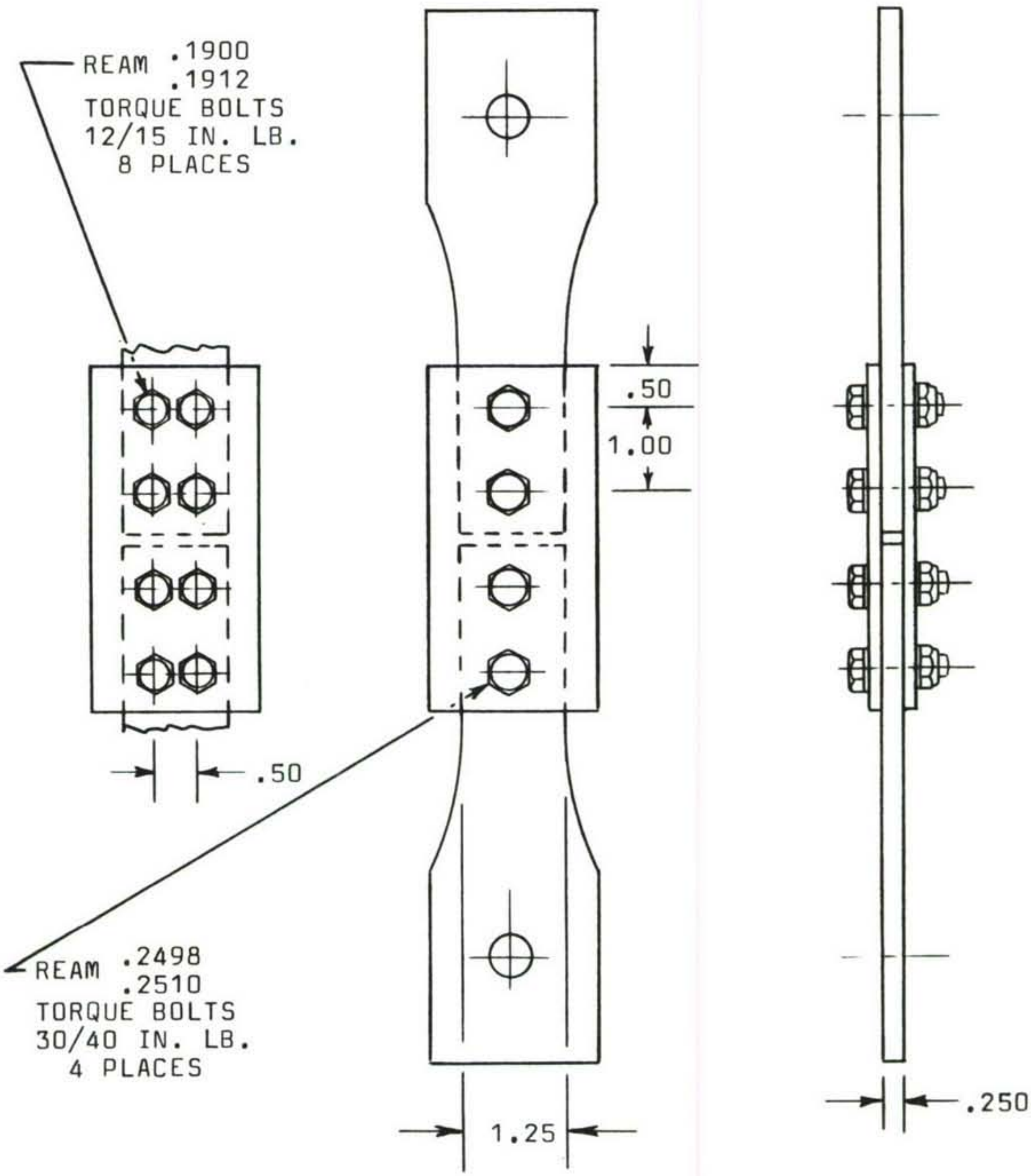


Figure 3. 4 and 8 Bolt Joint Specimen Designs.

Midway in the testing program an 8 Bolt Joint specimen was designed. The design requirements for this specimen included the exact duplication of mating surfaces of the A-26A critical wing area, and a closer duplication of fastener size and spacing. These requirements were added after analysis of the test results of the initial phases of the program indicated that specimen fatigue life might be significantly influenced by the presence of clad material on fretting surfaces, and by the size and number of bolts transferring load.

All aluminum parts designed prior to the 8 Bolt Joint design were made from 2014-T6 clad aluminum alloy plate. Steel parts were made from alloy steel plate, AISI 4130 Condition N, and all bolts were aircraft standard, NAS 1303, 1304, and 1305, with a heat treat of 160,000 psi minimum strength.

The 8 Bolt Joint design requires bare material to be used for the specimen and clad material for the splice plates. This duplicates the combination of materials used on the critical sections of the A-26A wing structure, with clad plate skin panels bolted to bare machined spar caps. The bolt size and the very close spacing are typical of that found on the wing spar caps, with 3/16 inch screws prior to the reaming process and 1/4 inch after the process has been accomplished.

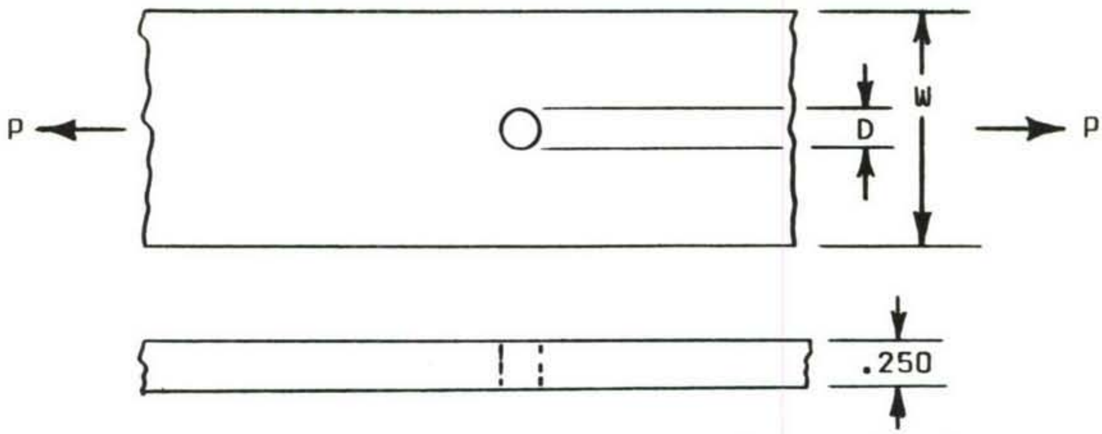
Stress concentration factors and maximum stresses are computed for the open hole specimen designs. These values are taken from Reference (6). For symmetrically located holes,

$$f_{net} = \frac{D}{(W - D)t}$$

$$K_t = \frac{f_{max}}{f_{net}}$$

$$f_{max} = K_t \ f_{net}$$

where f_{net} is computed from dimensions of the specimen designs, and K_t is taken from the plot of Reference (6), page 84.



D	W	D/W	K_t	f_{net}/P	f_{max}/P
.125	1.25	.100	2.72	3.56	9.65
.187	1.25	.150	2.61	3.76	9.80
.312	1.25	.250	2.43	4.26	10.38
.437	1.25	.350	2.30	4.92	11.32

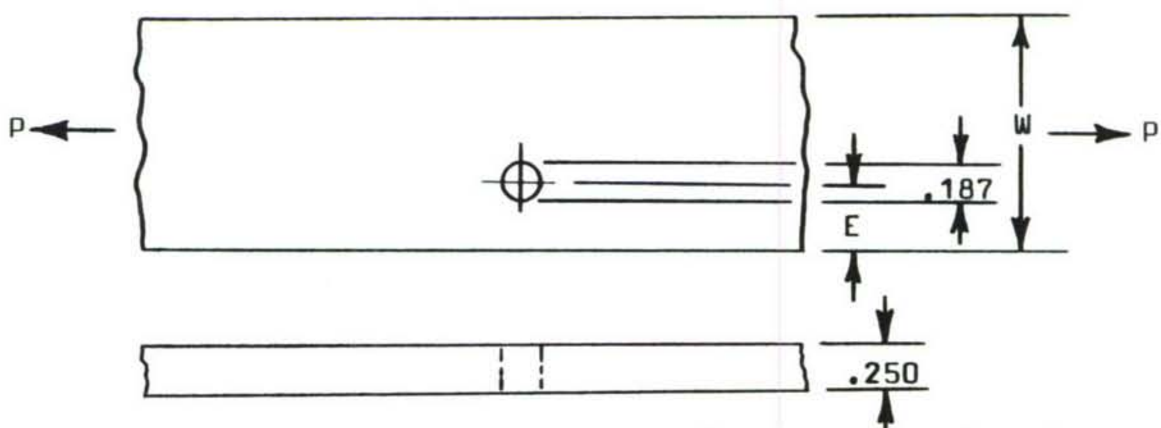
For unsymmetrically located holes, Reference (6), page 86, Gross Area Stress,

$$f = \frac{P}{Wt}$$

Maximum Stress at the Edge of the Hole,

$$f_{max} = \frac{f_{max}}{f} \times f$$

where $\frac{f_{max}}{f}$ and K_t are the functions of E/D and $\frac{W}{E} - 1$.



E	E/D	D/(2E)	(W/E)-1	f_{max}/f	K_t	f_{max}/P
.467	2.50	.200	1.67	3.14	2.51	10.07
.374	2.00	.250	2.34	3.21	2.44	10.30
.280	1.50	.333	3.45	3.38	2.36	10.82

Prior to the start of the specimen fatigue testing, qualification tests were run on Loaded Hole and 4 Bolt Joint specimens. These tests consisted of applying cyclic loads with negative minimum loads (compression loading) to verify the strength and stability of the loading fixtures and specimen combination. These specimens had strain gages installed so that the individual bolt loads could be computed.

From the strains recorded during the qualification testing the ratio of load taken by the steel doubler plates of the Loaded Hole specimen was computed as .284. Because of the closer bolt spacing and wider splice plates of the 4 Bolt Joint specimens, it was not possible to compute a set of loads from the strain readings of the Joint specimens. It was assumed, after an investigation of the strain readings, and noting that no premature failures occurred in the splice plates, that the two bolts transferring load from one piece of the specimen to the splice plates were equally loaded. This assumption was also used for the 8 Bolt Joint specimen.

The ratio of bolt bearing stress to net tension stress:

$$\text{Loaded Hole:} \quad \frac{f_{br}}{f_{net}} = \frac{.284 \times 1.00}{.250} = 1.14$$

$$\text{4 Bolt Joint:} \quad \frac{f_{br}}{f_{net}} = \frac{1.25 - .25}{2 \times .25} = 2.00$$

$$\text{8 Bolt Joint:} \quad \frac{f_{br}}{f_{net}} = \frac{1.25 - (2 \times .187)}{4 \times .187} = 1.17$$

Specimens were fabricated using normal aircraft shop procedures. The location and drilling of bolt holes was more closely controlled than would normally be required for an airplane wing. This was necessary to insure consistent load transfer for each bolt, particularly for the Loaded Hole specimens, where the doubler plates could not be loaded without accurately located holes. One set of 4 Bolt Joint specimens (test 41 (b)) was tested with the standard sheet metal diameter of .253 inches for a 1/4 inch bolt. The test results were unchanged by the additional hole tolerance (Table LI).

Splice and doubler plates were each taken from one sheet of material, as were the .125 inch thick specimens. Four sheets of .250 inch thickness were used, three clad and one bare. Of the three clad sheets, Loaded Hole and Joint specimens were made primarily from sheets marked red and blue, while the Open Hole specimens were mainly from the sheet marked yellow, with some

additional specimens made from the sheet marked blue.

Tensile tests were run from samples taken from all the sheets used except for the steel, and all properties were well above the minimum guaranteed. (Reference Table 1)

Source inspection and net area data measurements were combined, as each critical dimension for each specimen was measured and recorded at the testing laboratory.

TABLE I

TABULATION OF STATIC TENSILE PROPERTIES

Specimen Number	Sheet Ident.	Coupon Width (in.)	Ultimate Stress (lb/in^2)	Yield Stress (lb/in^2)	Effective Length (in.)	Elongation (per cent)
517	Bare	1.25	69,000	64,000	2.0	15
518			68,000	63,000		16
519			69,000	64,000		15
735	Blue	1.25	69,800	65,000	2.0	16
736			69,800	64,900		15
737			69,500	64,700		16
738			69,900	64,700		17
739			69,200	64,400		18
740			69,000	64,100		16
741			69,500	64,600		18
901	.125	1.25	69,700	63,400	2.0	14
902			70,200	63,500		15
903			69,600	62,800		15
59	Yellow	.25	66,500	61,600	1.0	12
60			68,200	62,900		13
395			67,600	61,900		14
425			68,000	63,000		14
450			67,200	62,400		14
Q-4	Red	.25	67,600	62,800	1.0	13
562			68,400	63,000		13
721			68,200	62,700		13
Q-		.25	68,600	62,700	1.0	13
782			68,400	62,700		13
787			68,700	63,300		13
970	Bare	.25	68,200	61,900	1.0	13
993			67,100	60,400		12
966			67,300	61,500		13
966			68,500	63,400		13

Note (1) Yield Strength determined by .2 per cent offset.

Note (2) Coupons of .25 inch width were taken from failed fatigue specimens.

Note (3) Material properties from MIL-HDBK-5A, Reference (12),

	Clad Plate	Bare Plate	Clad Sheet
Ultimate Stress, F_{tu}, psi	63,000	66,000	65,000
Yield Stress, F_{ty}, psi	58,000	60,000	58,000
Ultimate Elongation, e, per cent	8	7	8

2. Testing Procedure

Krouse direct stress fatigue testing machines were used for specimen fatigue testing. A two-to-one load amplification mechanism was used as shown in Figure 4. Maximum load capacity for each of the two loading stations on each machine is 5,000 pounds direct, tension or compression, and 10,000 pounds at the amplified loading station. The amplification mechanism was designed for minimum loads greater than zero (tension-tension). A loading fixture for tests which had requirements for minimum loads less than zero (tension-compression) was designed to provide end fixity, or compression stability, to the specimens and was loaded by both stations of one machine simultaneously. This provided 10,000 pound capacity for either tension or compression.

Loading levels were set and monitored by use of an Ellis bridge-amplification unit coupled with a Tektronix oscilloscope. Maximum and minimum loading levels were set as shown in Figure 5. Cyclic loads were then monitored by observing the coincidence of the peaks with the pre-set values. The cyclic loading signal was provided by a load cell located in series with the test specimen with an electrical circuit connection to the Ellis unit.

Mean load was maintained automatically as permanent deformation occurred in the specimens. Hydraulic fluid was supplied to the mean load cylinder upon a signal from a limit switch. Limit switches were located with respect to the loading beam of the Krouse testing machine such that a change in beam bending deflection due to reduced loading would close one of the circuits.

Normal loading rates ranged between 1,150 and 1,500 cycles per minute. Tests with relatively high loading and low cycles to failure were reduced to 500 cycles per minute for improved cycle recording accuracy.

The temperature increase of specimens due to cyclic loading was less than ten degrees Fahrenheit.

Specimen loading grips are shown in Figure 6. The center bolt fits into the .500 inch diameter hole in the specimen, while the two bolts located on either side provide additional friction forces between the grips and the specimen.

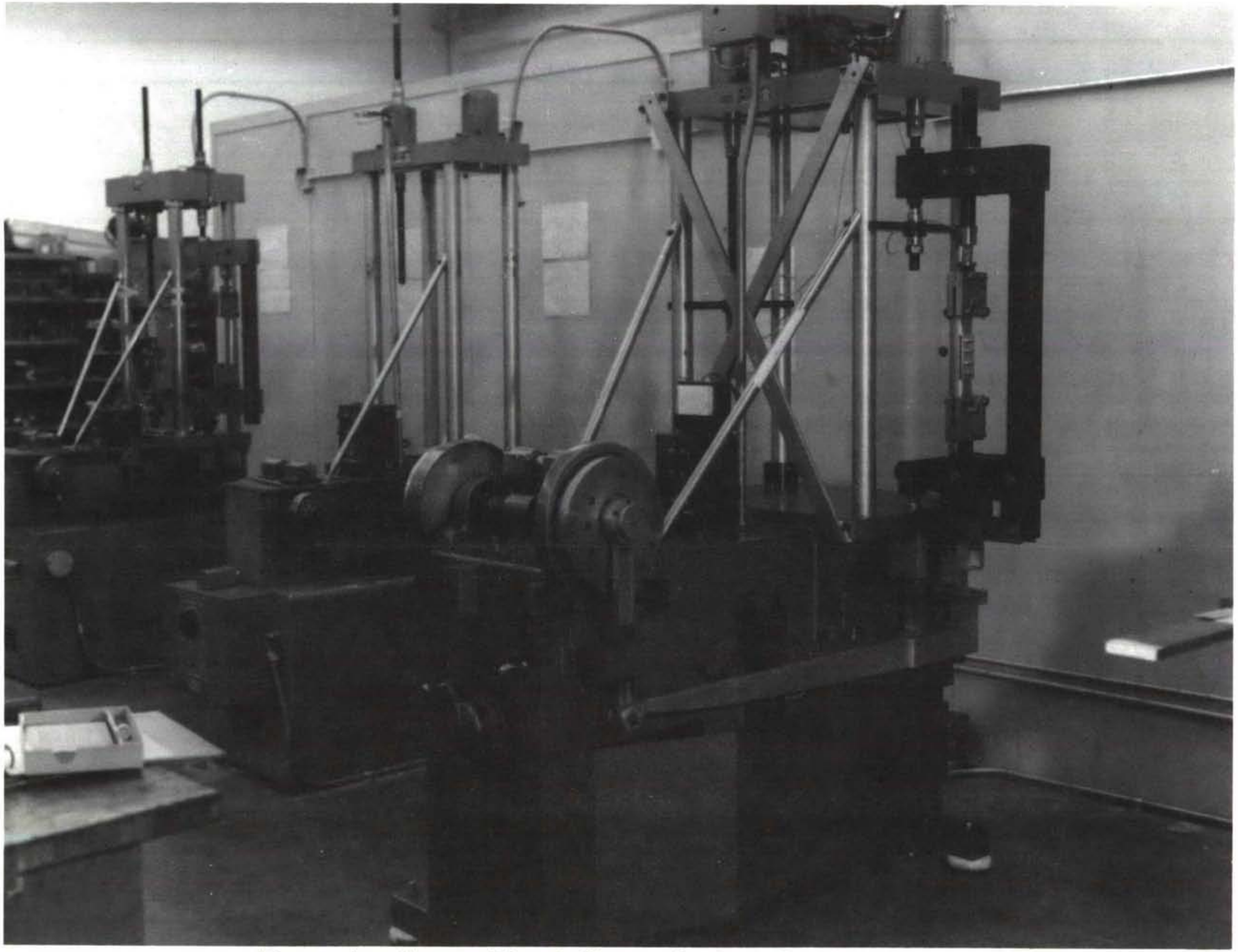


Figure 4. Krouse Direct Stress Fatigue Testing Machine with 2:1 Load Amplification Mechanism Installed.

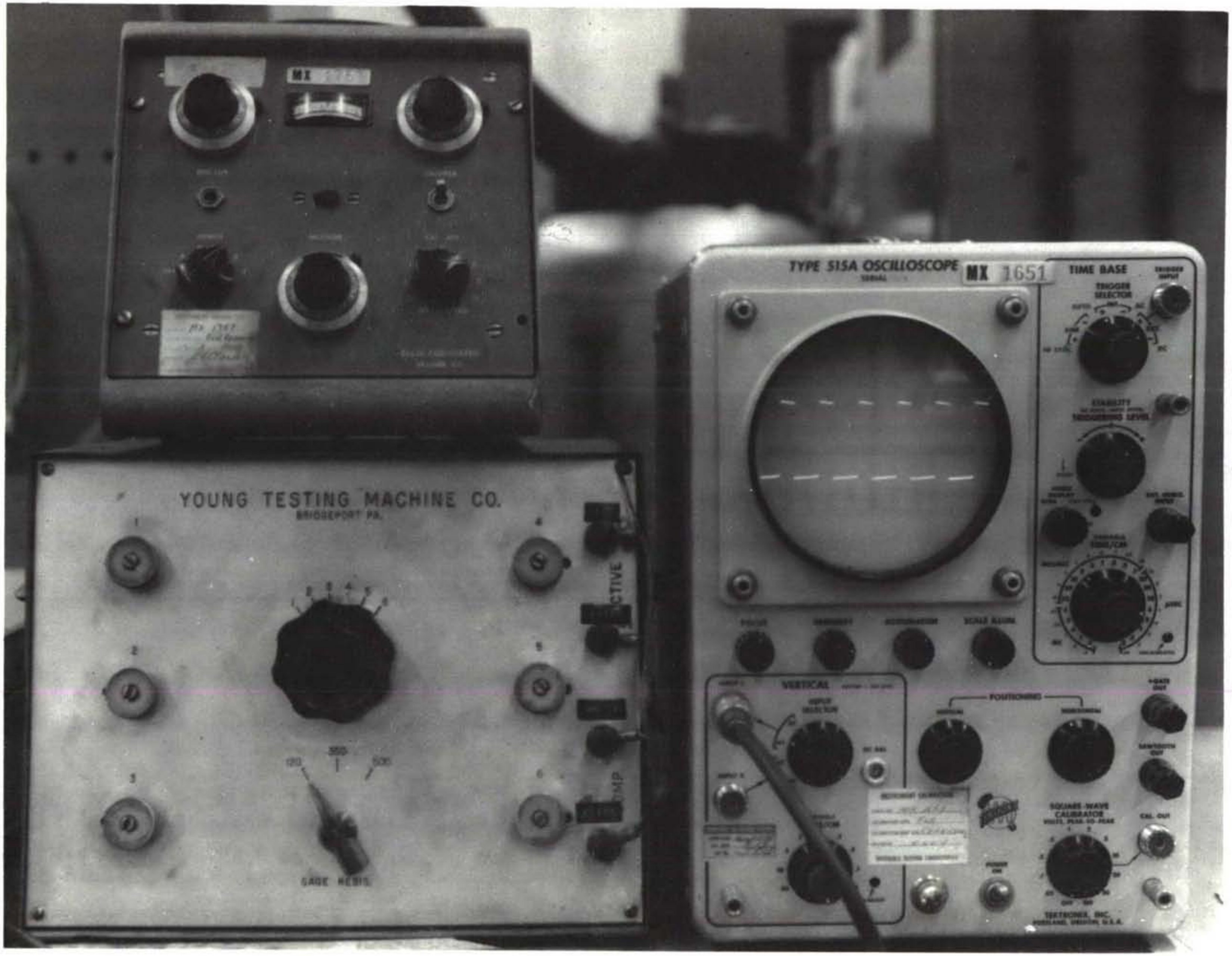


Figure 5. Ellis Bridge-Amplifier Unit, Young Selector and Balance Unit, and Tektronix Oscilloscope Used for Monitoring Cyclic Test Loads.

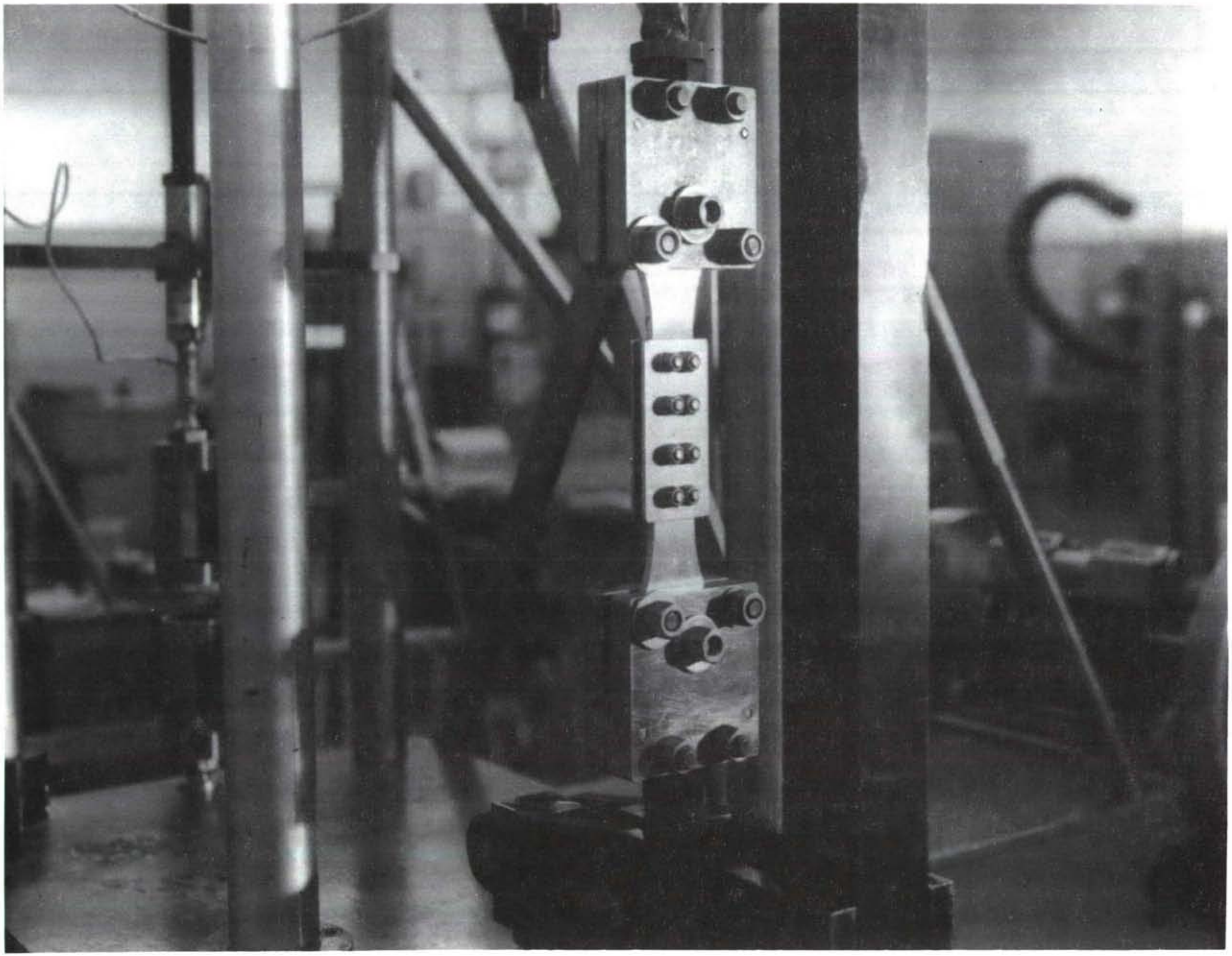


Figure 6. 8 Bolt Joint Specimen Installed in Grips.

3. Crack Propagation Rate and Residual Strength

Crack detection wires were used to determine the number of test cycles producing initial crack. The relationship between initial crack cycles and final failure cycles was evaluated so that test results based upon ultimate failure can be applied to analysis of structures where initial crack is the significant failure criterion.

Each crack detection wire consists of a length of .004 inch diameter enamel insulated copper wire. This wire forms a portion of a relay circuit, designed to shut off power to the fatigue machine upon failure of the wire. The wire is cemented across the area where a crack is expected to occur so that a crack in the structure causes a failure of the wire.

A number in excess of 100 crack wires were installed on open hole specimens, from which 86 successful readings were recorded. Of these, 32 were used in the reaming tests. 54 readings were taken for the purpose of comparing initial crack to final failure. (Table II)

It has been concluded that final failure results are adequate for reporting and drawing conclusions for the specimen testing program.

52 of the 54 initial crack cycles measured were more than 90 per cent of the comparable failure cycles, and 44 initial crack cycles were greater than 95 per cent of the failure cycles. Initial crack cycles is the number of load application cycles at which crack wire failure occurs. The crack wire is located 1/16 inch from the edge of the hole, with the crack propagating to a point slightly beyond the wire before producing sufficient strain in the wire to cause fracture.

The crack detection wire installation consists of the following procedures:

(a) The surface of the specimen is cleaned and a coat of air-drying acetate cement is applied to the area of the wire installation.

(b) The wire is placed in position, using a circular template, and a second coat of cement applied.

(c) The ends of the wires are stripped and soldered to terminal strips which provide connecting points for the wiring which completes the circuit from the specimen to the shut-off relay.

TABLE II

TABULATION OF CYCLES TO INITIAL CRACK

		N	N_c	$N-N_c$	N_c/N			N	N_c	$N-N_c$	N_c/N
TEST NUMBER	SPECIMEN NUMBER	CYCLES TO FAILURE	CYCLES CRACK WIRE FAILURE	CYCLES	PER CENT	TEST NUMBER	SPECIMEN NUMBER	CYCLES TO FAILURE	CYCLES CRACK WIRE FAILURE	CYCLES	PER CENT
1	910	221,000	215,800	5,200	98	6	267	106,700	102,900	3,800	96
2	70	84,500	81,100	3,400	96	6	268	111,400	110,600	800	99
2	71	57,500	49,400	8,100	86	6	269	39,400	38,800	600	98
2	72	46,600	44,200	2,400	95	6	270	88,500	85,400	3,100	97
2	73	27,100	25,400	1,700	94	6	271	37,800	37,700	100	99
2	74	9,700	9,000	700	93	6	272	34,100	34,000	100	98
3	9	66,400	61,400	5,000	93	6	273	12,200	12,100	100	99
3	10	73,600	69,400	4,200	94	6	274	12,500	12,300	200	98
3	11	57,600	57,000	600	99	7	219	77,800	74,900	2,900	96
3	12	41,100	39,200	1,900	95	7	220	94,700	87,400	7,300	92
3	13	18,400	18,100	300	98	7	221	38,900	37,800	1,100	97
3	14	19,300	18,900	400	98	7	222	46,600	46,100	500	99
3	15	9,200	9,000	200	98	7	223	13,400	13,200	200	98
3	16	8,300	8,100	200	98	7	224	16,500	16,300	200	99
4	316	96,700	92,300	4,400	95	7	225	6,200	6,100	100	98
4	317	61,000	59,900	1,100	98	7	226	8,500	8,400	100	99
4	318	62,500	62,200	300	99	8	235	76,400	75,200	1,200	98
4	319	25,100	24,900	200	99	8	236	201,400	195,700	5,700	97
4	321	10,100	8,000	2,100	79	8	238	47,000	46,100	900	98
4	322	9,400	9,300	100	99	8	239	16,500	16,300	200	99
5	300	143,200	140,700	2,500	98	8	240	21,500	21,100	400	98
5	301	59,300	58,200	1,100	98	8	241	9,900	9,100	800	92
5	302	39,400	38,600	800	98	8	242	11,400	11,300	100	99
5	305	22,300	22,000	300	98	9	243	100,500	92,300	8,200	92
5	306	20,300	20,200	100	99	9	244	101,400	98,100	3,300	97
5	303	9,500	9,400	100	99	9	245	48,600	46,500	2,100	96
5	304	11,300	11,200	100	99	9	246	45,500	42,700	2,800	94

Crack propagations of the specimens were compared to the data plotted in References (4) and (5), but no real correlation was noted. The basic reason for this is that the specimens are too narrow (1.25 inch) to produce a sufficient number of cycles during cracking to show any trend.

The method of Reference (4), based on test results of 7075-T6 alloy, indicates that for maximum stresses greater than 29,000 psi, the residual strength is exceeded at the time of crack wire failure. In every case the 2014-T6 specimens withstood at least 100 cycles of loading between crack wire failure and final failure.

4. Reporting Methods and Statistical Analyses

The basic data requirement of this testing program was to construct, from test results, a series of S-N curves for the purpose of evaluating fatigue life improvement processes. The initial series of tests was scheduled for developing basic, or control data, the intermediate phase for evaluating the best processes for fatigue life improvement, and the final phase for producing data reflecting the improvements gained.

Each test consisted of evaluating the fatigue life, expressed in number of load application cycles, for 16 specimens. Four load levels were selected for each test with four specimens tested under identical conditions at each load level. From the mean values of each group of four specimens, four points were plotted and an S-N curve drawn. (Four tests for each point unless otherwise indicated on the curve.)

Midway in the testing program, the requirements were modified and certain tests were changed from an S-N curve requirement to a fatigue life versus bolt torque requirement.

An S-N curve was drawn for each test with four different load levels, based on ultimate failure of the specimens (Appendix III), and fatigue life versus bolt torque curves were drawn from the data produced in the constant load tests with the addition of individual points taken from the variable load tests (Figures 15 through 18).

The data from each test is included in Appendix II. The logarithm (base 10) of each specimen failure cycles was taken, the average for each group of four computed, and the anti-log taken and recorded as N (log mean). The sample standard deviation, s, in terms of log cycles, was computed for each group of four specimens as follows:

$$s = \left[\frac{\sum_1^n (\log N - \text{mean} \log N)^2}{n-1}\right]^{\frac{1}{2}} \qquad \text{(Reference (8) page 114)}$$

N is the number of load cycles to failure, and n is the number of individual test results being considered; in most cases, $n = 4$.

Maximum and mean loads for each specimen test and measured critical dimensions for each specimen were taken from the laboratory data. Stresses were computed for each set of four specimens. The average net area for all 16 specimens of a test is shown at the top of each data sheet (Appendix II).

To provide a systematic method for plotting S-N curves, the method of Weibull (Reference (7)) was used. Each curve was plotted using the following procedure:

(a) The endurance limit stress was estimated.

(b) A plot of log log $(TS - \overline{S})/(S - \overline{S})$ versus log log N was made. TS is the ultimate tensile strength for the material, with 70,000 psi used in all cases, $\overline{S}$ is the endurance limit stress, and S is the maximum cyclic stress for any one load level.

(c) If the four points plotted produce a net curvature, then a new endurance limit stress is selected and the points replotted.

(d) Upon achieving a plot of four points with no net curvature, a straight line is drawn through the points. This is done by "eye", approximating a least squares fit between the line and the four points.

(e) S is plotted versus N on semi-log paper using the values taken from the straight line on the log log $(TS - \overline{S})/(S - \overline{S})$ versus log log N plot. A typical plot of log $(TS - \overline{S})/(S - \overline{S})$ versus log N drawn on a log grid is shown in Figure 7.

Three probability plots were constructed, Figures 8, 9, and 20, to show statistically the effect of using N(log mean) for taking the average of cycles to failure data (Reference (8) page 118). If the log N distribution is statistically valid, a plot of log N versus probability of failure will produce a straight line on probability paper, or a bell shaped, Normal, or Gaussian curve when plotted on coordinate graph paper. Because of the symmetry of the normal distribution, the mean and median coincide if the distribution is truly normal.

A composite set of data was compiled from tests 2, 5, 6, 9, and 8. The geometrical variations of the specimens used in these tests were minor, with the fatigue life as a function of net stress being reasonably consistent. Minor variations in maximum stresses at each load level were normalized by applying a factor to the fatigue life as derived from the S-N curve for test 2 (Figure 32).

The highest load level, 38,000 psi maximum cyclic stress, was plotted as a comparison for data taken from peened specimens, Figure 20, while the next lower load levels, with maximum stresses of 33,000 psi and 28,900 psi, Figures 8 and 9, were plotted as a check on the distribution only.

In all three cases, the mean values are higher than the median values. The second load level (Figure 8) data was relatively normal along with the data for peening (Figure 20). The highest and third highest loadings (Figures 20 and 9) produced plots

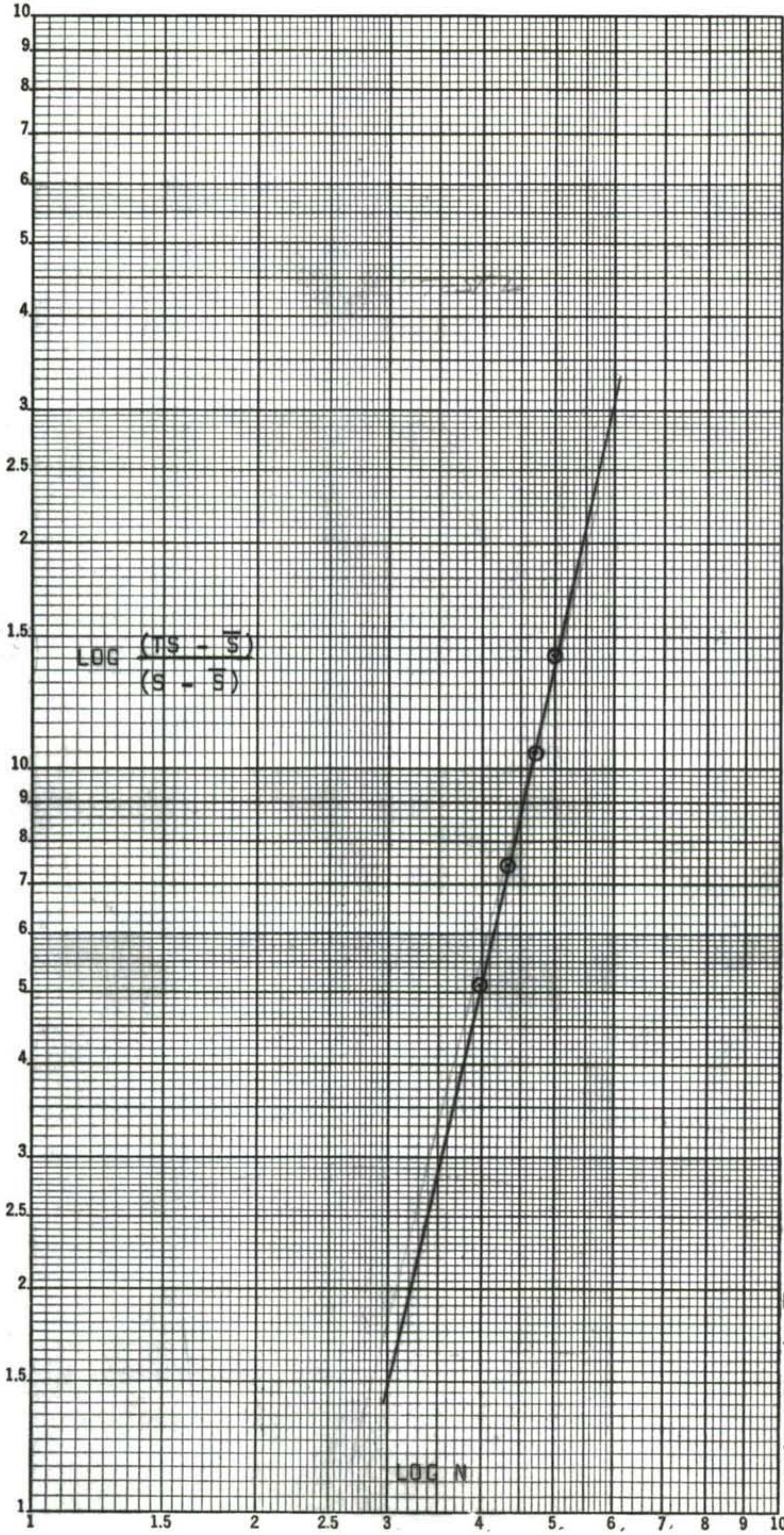


Figure 7.

Typical plot of Log $\frac{(TS-\overline{S})}{(S - \overline{S})}$ versus log N. (Test Number 3)

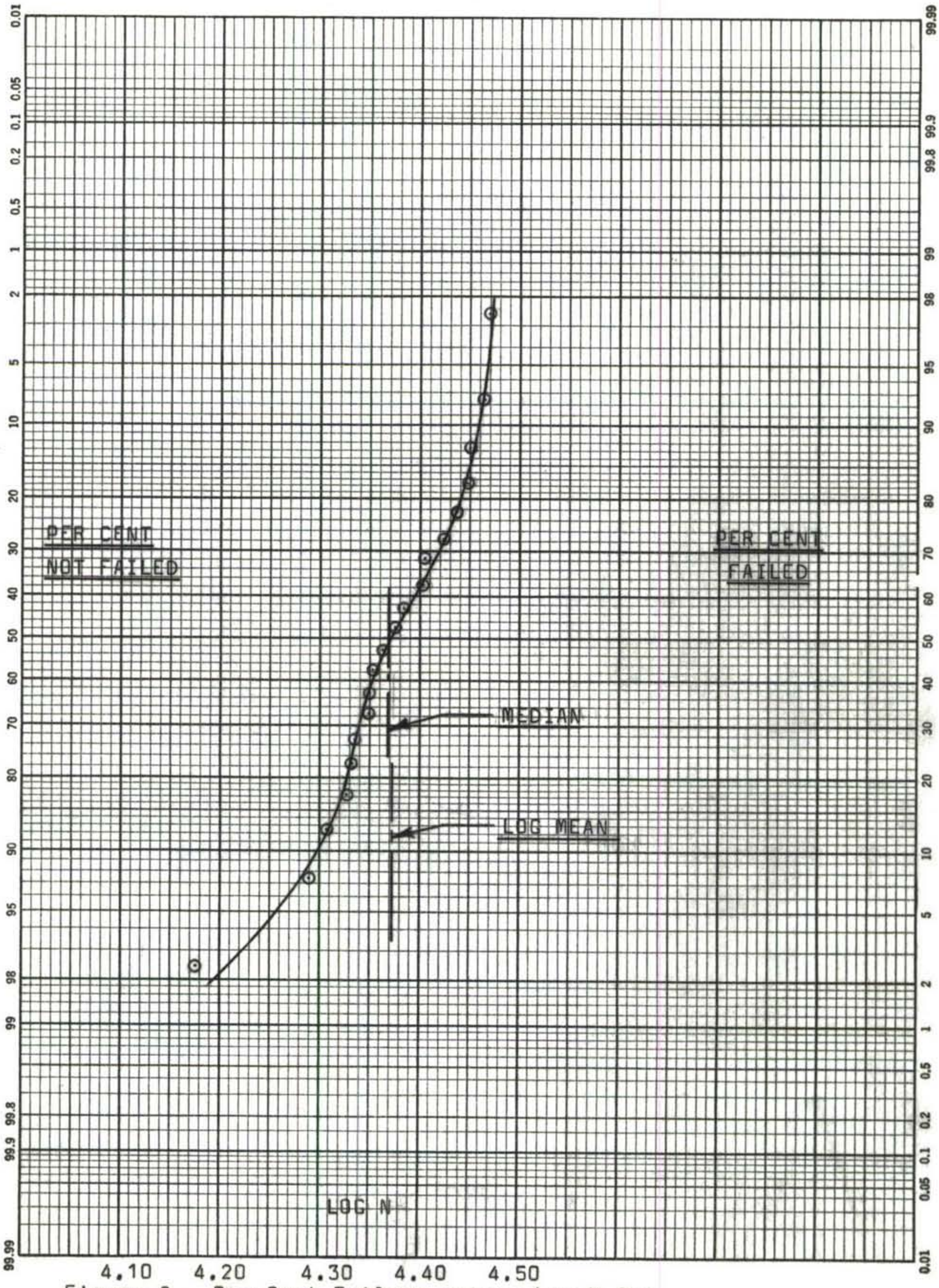


Figure 8. Per Cent Failure versus Log N for Tests 2, 5, 6, 9, and 8. Second Load Level, Maximum Stress, 33,000 lb/in^2.

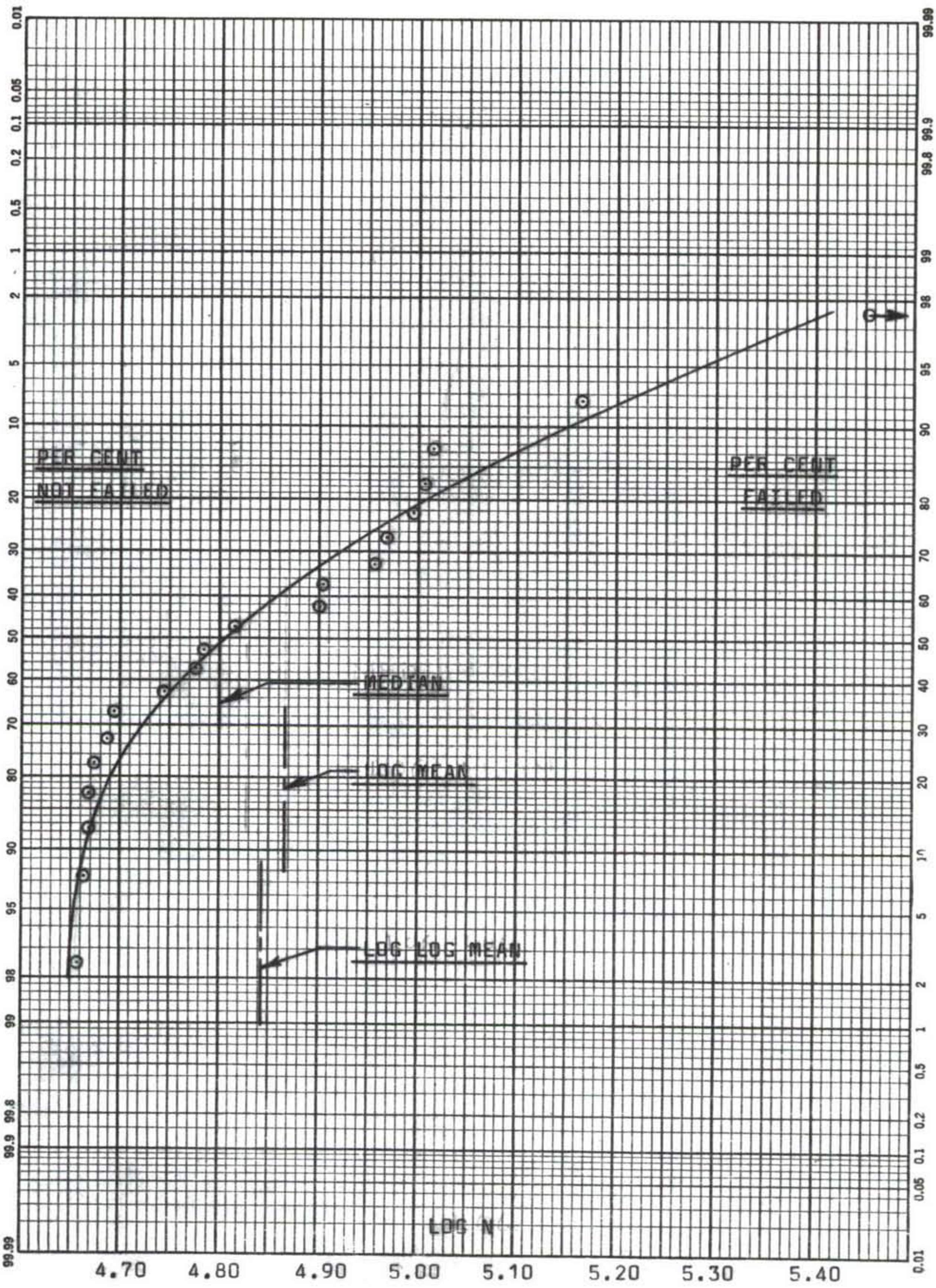


Figure 9. Per Cent Failure versus log N for Tests 2, 5, 6, 9, and 8. Third Load Level, Maximum Stress, 28,900 lb/in^2.

with a definite curvature, negative second derivative, with the mean values somewhat higher than the median values. A shift in one or two data points in the area of 50 per cent failure would shift the median somewhat; however, the curvature is quite pronounced and would not disappear completely due to a shift in the median data. An investigation of these distributions indicates that log log N would plot more closely to a normal distribution than log N. Because only 20 points were used for each plot, and these were taken from several different tests, the evidence is not conclusive and the log N distribution, being the most widely accepted method for reporting fatigue failure cycles, has been used in this report.

5. Chronology of the Program

The original testing program requirements consisted basically of three phases. The first phase consisted of a series of control, or basic S-N curve, tests for different specimen configurations. The second phase was to be exploratory, or a process optimization program; while the final phase was to be a duplication of the first phase except that the processes optimized in the second phase would be included.

The first phase and approximately 50 per cent of the second phase were completed as scheduled. Midway into the second phase, serious doubts were raised as to the feasibility of optimizing the processes within the confines of the existing testing schedule. The following problems were evident at this point:

(a) While both of the required reaming processes were evaluated as scheduled for the Open Hole tests, Loaded Hole and 4 Bolt Joint tests with these processes applied produced a wide range of results.

(b) A test of peened Open Hole specimens did not produce any increase in fatigue life.

(c) Tests of peened Loaded Hole and 4 Bolt Joint specimens produced inconsistent results, with some specimen tests indicating no improvement and the remaining tests indicating significant improvement.

(d) 4 Bolt Joint specimens were severely pitted in the surface area adjacent to the holes, with some instances of splice plates and specimens being welded together at points just fore and aft of a bolt hole (Figures 10 and 11). This condition was evident at the time of disassembly for reaming, after initial load cycling had been applied.

It was concluded that in order to evaluate the effects of reaming and peening and to produce meaningful results, the effects of surface friction and bolt preloads would have to be evaluated. Also, it was concluded that variations in the peening process should be investigated in an effort to find a system that would produce an improvement in fatigue life.

Subsequently, the remainder of the testing schedule was changed as follows:

(a) Additional Open Hole specimens were scheduled for various methods of peening prior to fatigue testing.

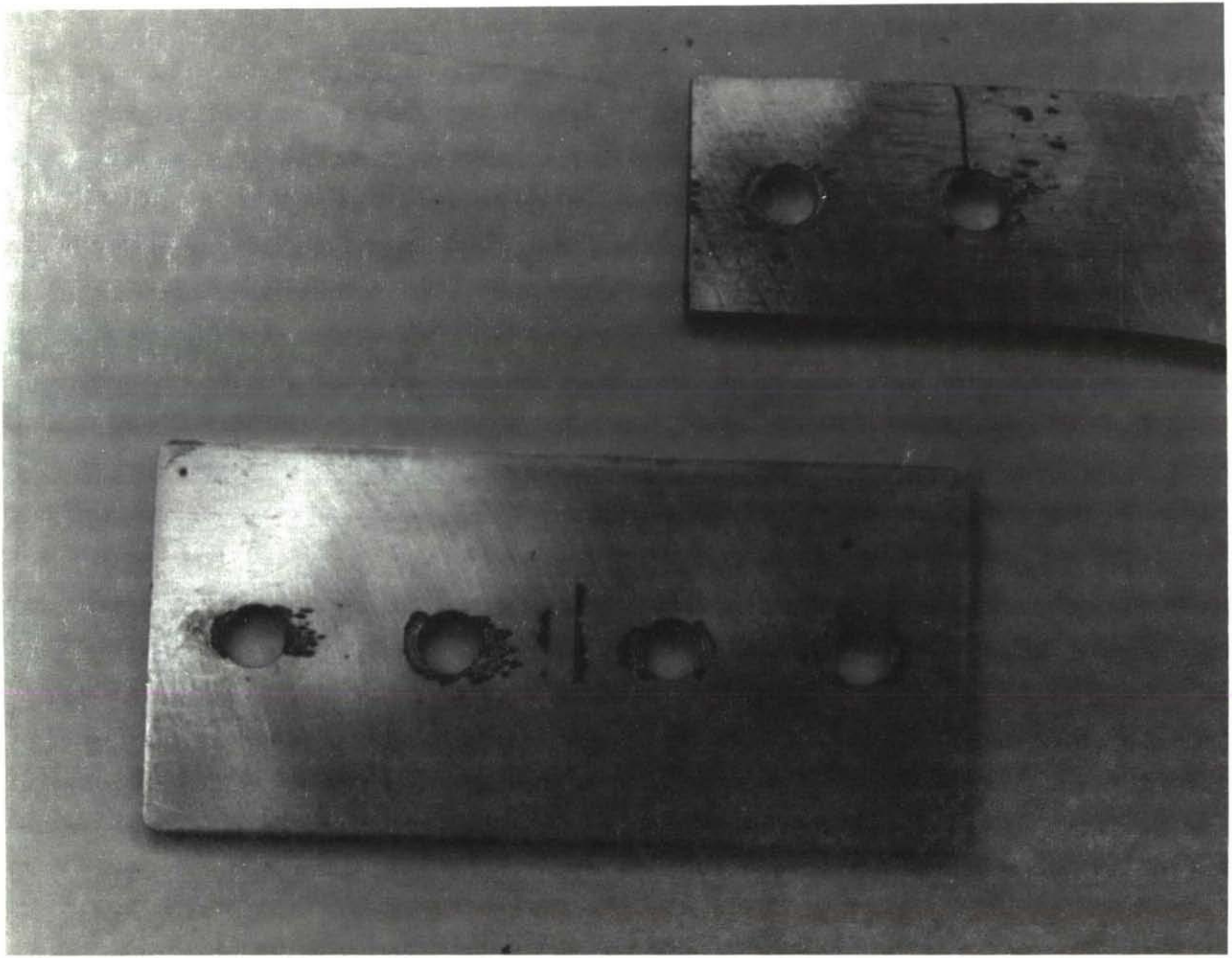


Figure 10. Failed 4 Bolt Joint Specimen and Splice Plate. Pitting Around Holes and at End of Part is Due to Fretting.

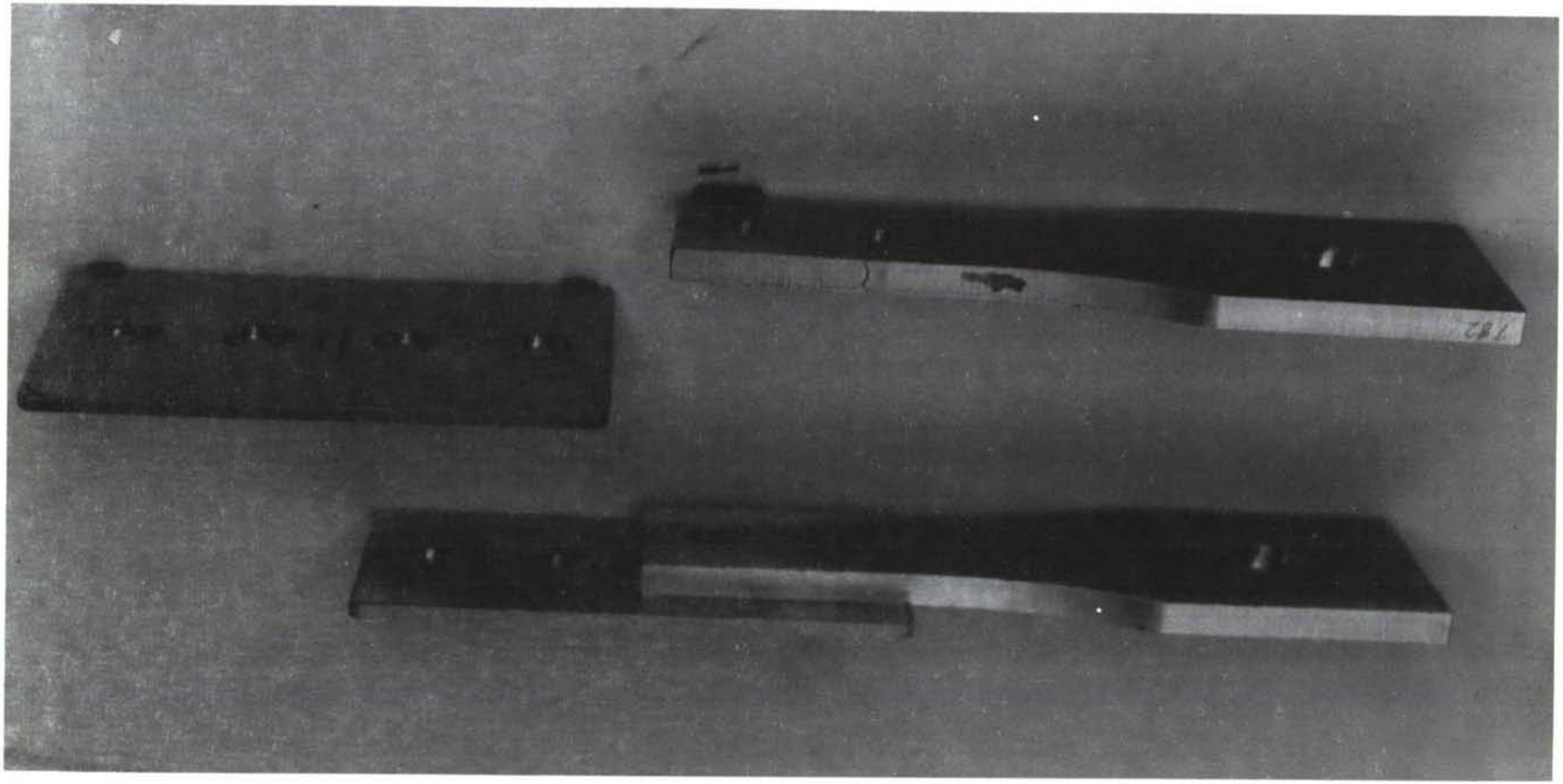


Figure 11. Failed 4 Bolt Joint Specimen and Splice Plate Welded Together Due to Fretting.

(b) Loaded Hole and 4 Bolt Joint specimens were scheduled for constant load tests with variation in bolt preload. These tests included reaming, peening, and combinations of both, as well as basic fatigue testing.

(c) A new joint specimen was designed. The possibility that surface material and bolt preloads could exert a strong influence upon fatigue life made it desirable to test a specimen which duplicated the conditions of the critical sections of the A-26A wing even more closely than the original Loaded Hole and 4 Bolt Joint designs.

(d) Since the conclusions as to the effect of reaming on an Open Hole specimen were well established, it was decided to test only two additional sets of Open Hole specimens with the reaming process applied. These specimens had small hole edge distance, and two reaming processes were applied during testing.

The remaining portion of the testing was completed essentially as rescheduled. The peening process was investigated and it was concluded that no significant increase in fatigue life could be attributed to peening.

The effects of bolt preload and surface material were isolated. The effect of the preload was found to be the significant factor for producing large variations in fatigue life. Plots were made of fatigue life versus bolt torque for unprocessed specimens, and for specimens which had peening and reaming processes applied. Once the effect of bolt preload was evaluated, it was possible to explain the variations in the original tests of peened Loaded Hole and 4 Bolt Joint specimens. It was concluded that because these specimens had been assembled after the peening had been applied and no bolt torque measurement was made, the variation in life was due to variation in bolt preload rather than the influence of the peening process.

The effect of the reaming process was evaluated after conclusions had been made concerning the bolt preload effects. This evaluation was reasonably conclusive despite some areas where analytical evaluations became somewhat indeterminate because of the large increase in fatigue life due to the effects of bolt preload.

6. Problem Areas

Two basic problem areas were encountered in the process of scheduling and evaluating the fatigue testing program. One of these was the evaluation of the effect of reducing fatigue damage around a hole by removing material (reaming process), and the other was the variations of specimen fatigue test results encountered when the testing was done during different time periods.

The problem of making a quantitative evaluation of the effect of the reaming process is basically the result of the uncertainties inherent in fatigue testing in general. There being no way of determining what the life of a specimen would be had it not been reamed, it was necessary to use average data for making comparisons. The evaluation of the Open Hole reaming tests did not present any unexpected difficulties, but the Loaded Hole and Joint specimen tests proved to be somewhat more difficult to evaluate.

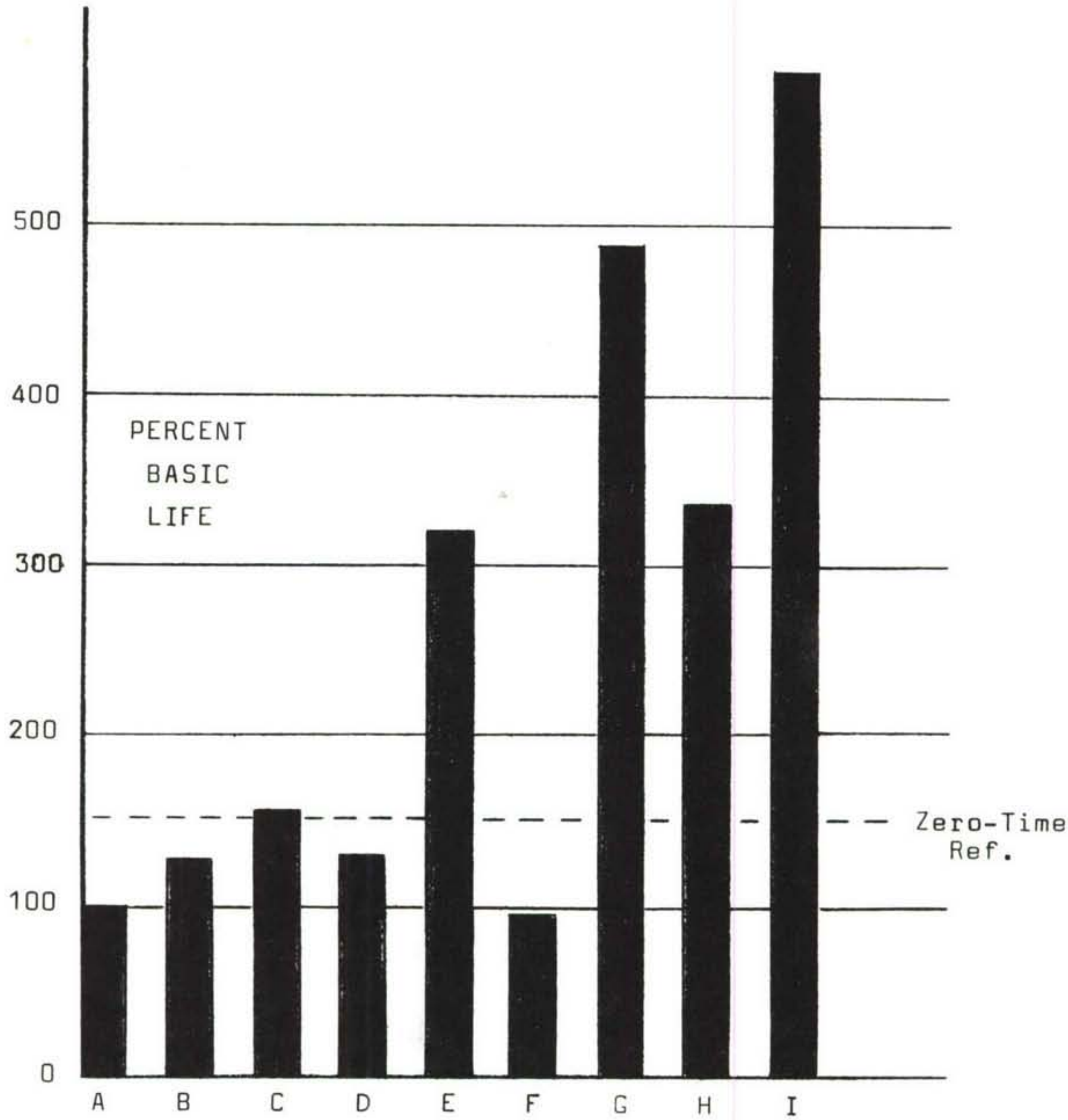
The increase in life due to the installation of larger bolts during the reaming processes applied to Open Hole and 4 Bolt Joint specimens was so pronounced as to make the evaluation of the reaming process almost indeterminate. In addition, these specimen tests did not always produce S-N curves of the classic shape as did the Open Hole specimens. On occasion, a middle stress range test would produce a relatively long average life, which tends to make the S-N curve approach a straight line.

The reaming process for initial damage values of greater than 60 per cent of failure had to be evaluated mainly by deduction, as the number of failures or incipient cracking occurring after 60 per cent of average life reduced the number of successful tests to the extent that explicit evaluations were not possible.

The basic total effect of the reaming process is, however, quite pronounced. The areas where the combination of reaming and bolt size produce definite improvements in fatigue life have been defined and specific confirmation of the process for the A-26A airplane has been achieved by virtue of the 8 Bolt Joint tests.

Some problems were encountered in producing consistent test results over a period of time, although not to an extent that would alter the test results. Occasionally, identical specimen tests run during a given time period would produce consistent results, but of a considerably different magnitude from comparable tests run during a different period of time. The causes of these discrepancies could be the subject of further investigation.

This effect was not evident in the A-26A wing fatigue testing, where three different wings were used (two lefts and one right) and the failures were quite consistent. The wing test was run over a time period of several months and ambient effects such as temperature and humidity averaged out over the period. Also, it is possible that mixed load cycle testing is more consistent than single load level testing.



A = 100% Life Ref.

B = Open Hole Reaming, .03 in., 25-66% Basic Life.

C = Open Hole Reaming, .06 in., 25-66% Basic Life.

D = Loaded Hole Reaming, .03 in., 50% Basic Life.

E = Loaded Hole Reaming, .06 in., 50% Basic Life.

F = 4 Bolt Joint Reaming, .03 in., 50% Basic Life.

G = 4 Bolt Joint Reaming, .06 in., 50% Basic Life.

H = 8 Bolt Joint Reaming, .06 in., 50% Basic Life, Peened.

I = 8 Bolt Joint Reaming, .06 in., 50% Basic Life, Peened, Mixed Loading Cycles.

Note: All the bars extending above the dashed line are considered to be properly zero-timed.

Figure 12. Comparison of Fatigue Life Ratios of Different Specimen Designs and Different Reaming Processes.

7. Reaming

a. General

The reaming process as used for the purpose of gaining additional fatigue life for a structural member consists of reaming an existing hole to a slightly larger diameter. The dimension used to define the process is the difference in inches between the original hole diameter and the hole diameter produced by the reaming process. In the specimen testing program, two such differential dimensions are used, referred to as .03 inch ream, and .06 inch ream.

These ream sizes are used because of the bolt diameters available; the .03 inch ream coincides with a 1/32 inch oversize bolt, which has the same head, thread, and nut as the smaller nominal size, and the .06 inch ream for the next nominal size of bolt and nut.

These two reaming processes were used on the A-26A wing during the cyclic test program, and subsequently applied to the A-26A service airplanes as modification requirements.

Two modifications were designed and installed on the A-26A fleet aircraft as a result of information gained from the wing cyclic test program. After these modifications were installed, all the screw holes in critical areas of the wing were enlarged .06 inch and larger standard diameter bolts were installed.

b. Open Holes

The test results for specimens with reamed open holes are summarized in Tables III and IV. The damage reduction factors have been summarized and plotted in Figure 13. The fatigue life ratios are summarized in bar chart form in Figure 12.

Figure 14 shows the fatigue damage accumulation versus distance from the edge of a hole where a given number of stress cycles have been applied. This is a cross plot of data taken from Figure 13. The purpose of the reaming process is to remove material near the edge of the hole and leave only relatively low damage in the remaining material.

The damage reduction factor, $\Delta D/D_o$, is the ratio of damage removed to the damage that existed before reaming. The damage that remains after reaming is equal to the original damage, D_o, multiplied by the quantity, $(1-\Delta D/D_o)$. If $\Delta D/D_o$ is 1.00, the material is completely zero-timed, or all damage removed.

The damage reduction factors plotted in Figure 13 are derived from the data of Tables III and IV, and also from investigating

TABLE III

DAMAGE REDUCTION FACTORS FOR .03 INCH REAM

OPEN HOLE SPECIMENS

1	N-FAILURE	101000	29600	10380	107800	42200	12900	116500	39600	13700
2	N-REAM	20000	6000	2000	33000	10000	3300	43000	12900	4300
3	REAM, IN.	.03	.03	.03	.03	.03	.03	.03	.03	.03
4	TEST NO.	14(b)	14(c)	14(d)	15(b)	15(c)	15(d)	16(b)	16(c)	16(d)
5	MAXIMUM STRESS	28900	32500	38100	28600	32100	38000	28700	32100	37900
6	BASIC LIFE	70000	28000	10100	80000	30500	10200	78000	30500	10400
7	TEST NO. (FIG. 32)	2	2	2	2	2	2	2	2	2
8	LIFE RATIO $\boxed{1}/\boxed{6}$	1.443	1.058	1.028	1.347	1.386	1.263	1.492	1.300	1.319
9	AVERAGE LIFE RATIO									1.293
10	N-NET $\boxed{1}-\boxed{2}$	81000	23600	8380	74800	32200	9600	73500	26700	9400
11	NET LIFE RATIO $\boxed{10}/\boxed{6}$	>1	.843	.830	.934	>1	.941	.942	.875	.904
12	1.000 - $\boxed{11}$	--	.157	.170	.066	--	.059	.058	.125	.096
13	REAM DAMAGE $\boxed{2}/\boxed{6}$	.286	.215	.198	.413	.328	.324	.552	.423	.413
14	DAMAGE REDUCT. $\boxed{13}-\boxed{12}$	--	.058	.028	.347	--	.265	.494	.298	.317
15	DAM.RED.RATIO $\boxed{14}/\boxed{13}$	1.0	.27	.14	.84	1.0	.82	.89	.71	.77
16	AVE. DAMAGE REDUCTION									.72

NOTES:
(1) Values in rows 1, 2, 3, 5, are taken from the data sheets for the tests referenced in row 4. Values for row 6 are from the curves referenced in row 7.
(2) Ream damage, row 13, $D_o = n/N$, where n is ream cycles, and N is basic life.
(3) Damage reduction ratio, row 15, is damage reduction factor, the ratio of D, row 14, to D_o, row 13.

TABLE IV

DAMAGE REDUCTION FACTORS FOR .06 INCH REAM.

OPEN HOLE SPECIMENS

1	N-FAILURE	179000	39000	11700	143000	37800	14500	111000	48000	14600
2	N-REAM	20000	6000	2000	33000	10000	3300	43000	12900	4300
3	REAM, IN.	.06	.06	.06	.06	.06	.06	.06	.06	.06
4	TEST NO.	18(b)	18(c)	18(d)	19(b)	19(c)	19(d)	20(b)	20(c)	20(d)
5	MAXIMUM STRESS	28600	32200	37900	28600	32300	38100	28700	32400	38000
6	BASIC LIFE	80000	30000	10400	80000	29000	10100	78000	28000	10200
7	TEST NO. (FIG. 32)	2	2	2	2	2	2	2	2	2
8	LIFE RATIO $\boxed{1}$ / $\boxed{6}$	2.240	1.328	1.124	1.790	1.303	1.435	1.422	1.715	1.431
9	AVERAGE LIFE RATIO									1.532
10	N-NET $\boxed{1}$ - $\boxed{2}$	159000	33800	9700	110000	27800	11200	68000	35100	10300
11	NET LIFE RATIO $\boxed{10}$/$\boxed{6}$	>1	>1	.933	>1	.958	>1	.872	>1	>1
12	1.000 - $\boxed{11}$	--	--	.067	--	.032	--	.128	--	--
13	REAM DAMAGE $\boxed{2}$ / $\boxed{6}$	.250	.200	.192	.412	.345	.327	.551	.461	.422
14	DAMAGE REDUCT. $\boxed{13}$-$\boxed{12}$	--	--	.125	--	.313	--	.423	--	--
15	DAM.RED.RATIO $\boxed{14}$/$\boxed{13}$	1.000	1.000	.652	1.000	.907	1.000	.768	1.000	1.000
16	AVE. DAMAGE REDUCT.									.925

Reference Table III for general notes.

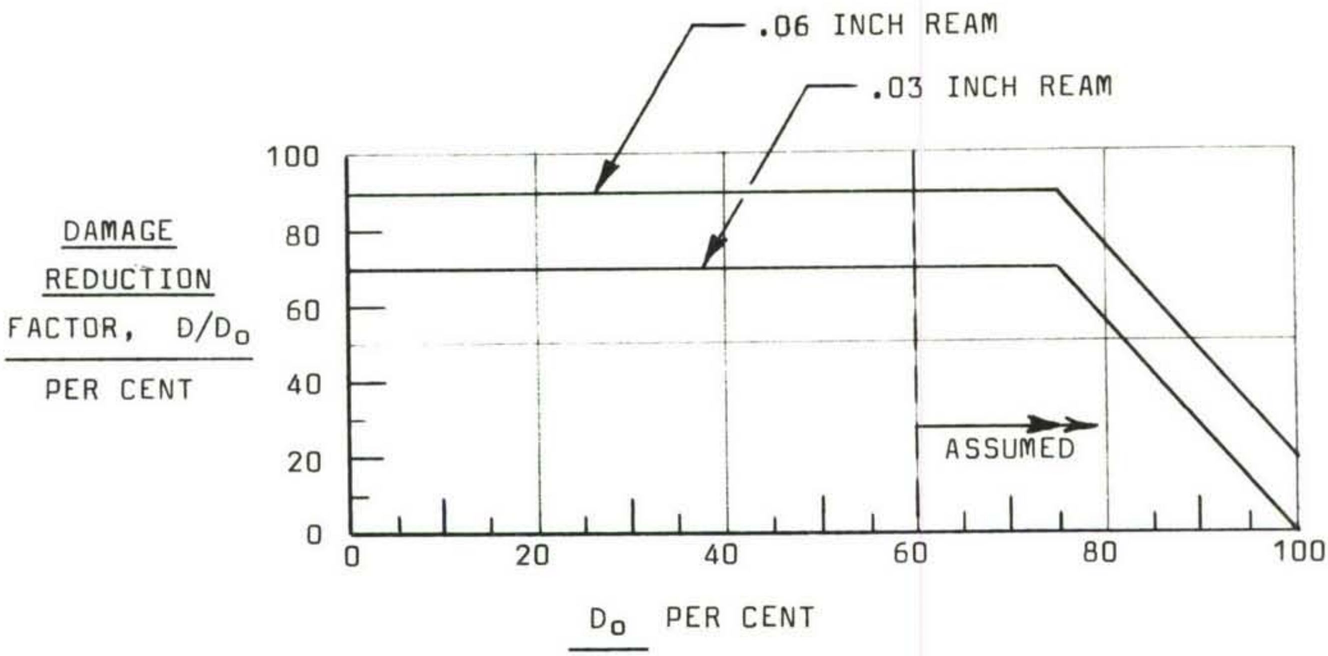


Figure 13. Open Hole Damage Reduction Factor versus Existing Damage for Two Reaming Processes.

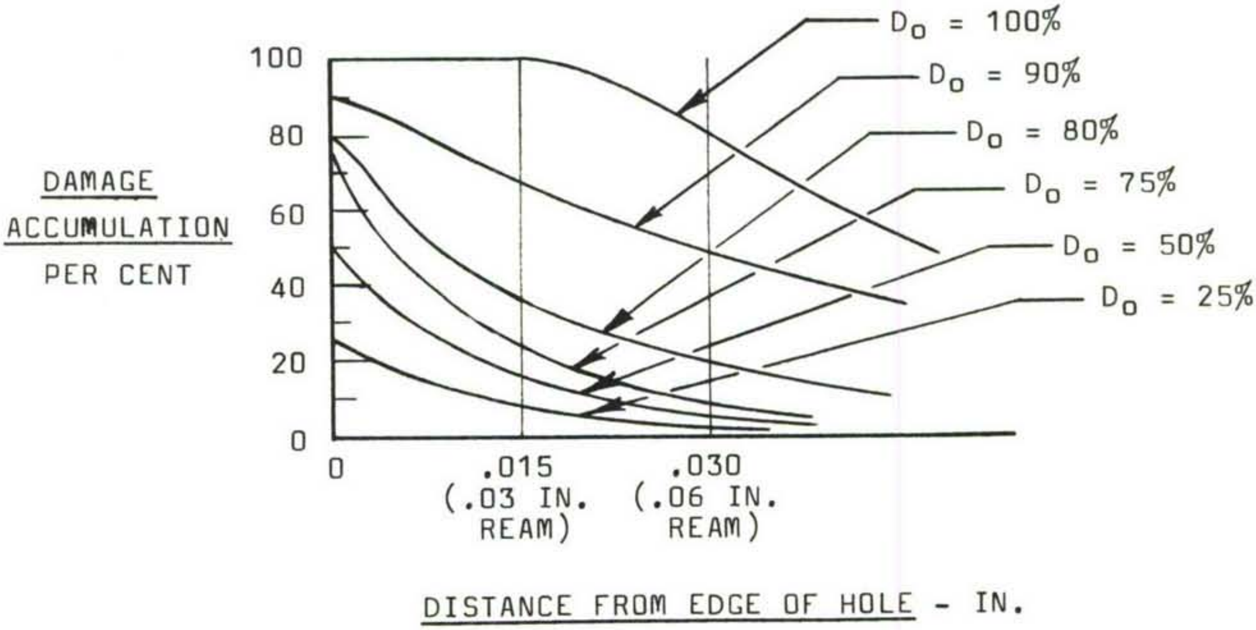


Figure 14. Fatigue Damage Accumulation versus Distance from Edge of Open Hole.

individual specimen test results from a number of tests. Because of the scatter inherent in any series of fatigue tests, and because it is impossible to prove explicitly the conditions that exist when fatigue cycling approaches the initial crack, or 100 per cent damage, the curve is drawn in a simplified form, and believed to be somewhat conservative for most values of existing damage. Reduction factors taken from this plot were applied to each individual test for the double ream tests, test numbers 50 and 40 (Tables XXVII and XXIX). An iteration process was used to solve for the basic fatigue life for each specimen. Basic fatigue life is the number of cycles to failure that would have been realized if the specimen had not been reamed. In all cases the results were reasonable and consistent.

The damage reduction factors are based upon the following considerations:

(a) The .06 inch ream was generally quite successful for zero-timing. A .9 damage reduction factor for existing damage up to 75 per cent is in most cases conservative.

(b) The .03 inch ream produced good results in most cases, but was definitely not as consistent as the .06 inch ream. A damage reduction factor of .7 is the average for the open hole reaming tests and is the number derived from the A-26A wing cyclic test. (Reference (1))

(c) The selection of 75 per cent existing damage for the change of slope is arbitrary and the reduction values for this area may be somewhat unconservative. While the validity of the reduction factors from zero to 60 per cent existing damage is reasonably well substantiated by test, the area from 60 per cent to 100 per cent existing damage is difficult to prove by test because of the high probability of failure before the reaming is applied. A more exact definition of this area is beyond the scope of this program.

c. Loaded Holes and Joints

The damage lines of Figure 14, and the damage reduction factors of Figure 13 apply to loaded holes and joints in a general way. Evaluation of fatigue life for parts in which critical holes are subject to bearing stresses and also to the effects of splice or doubler plates on the surfaces adjacent to the holes is much more complicated than for open holes. For this reason the damage reduction factors generally cannot be applied to loaded holes with the same confidence as applied to open holes.

Damage reduction factors for a .03 inch ream must be used with care. Reasonably good correlation for a damage reduction factor of .7 has been made for Loaded Hole specimens and for the A-26A wing fatigue test results. The 4 Bolt Joint specimen tests did not show any damage reduction due to this process. The lack of damage reduction due to a .03 inch ream applied to the 4 Bolt Joint specimen may be due to high bearing stresses, or to the friction or fretting effects of mating clad surfaces. A more detailed evaluation of these effects is beyond the scope of this program.

The damage reduction factor of .9 for a .06 inch ream is substantiated for both Loaded Hole and Joint specimen tests (Tables V and VII). Damage reduction values are difficult to determine accurately for .06 inch reams because the fatigue life is greatly increased by the preload of the larger bolts installed. The cycles run prior to reaming were a small fraction of the total, so that small per cent variations in total cycles produced large variations in damage reduction values.

Two specific test results should be explained:

(a) The failure cycles for Loaded Hole specimens reamed at 50 per cent life and with 70 inch-pounds torque is taken as 100,000 cycles. The test results include two specimens that failed between 30,000 and 40,000 cycles (Table XXXIX), and are considered to be not typical by comparison with other data. The 1.00 damage reduction factor for this point is assumed, and is not substantiated rigorously by test.

(b) The failure cycles for the 4 Bolt Joint specimen at 70 inch-pounds torque (Figure 17), is taken as 102,700 cycles. Two specimen tests of test number 43(d) had results that are considered untypically high, 160,000 and 347,900 cycles (Table LII). These were ignored in the plot of fatigue life versus bolt torque and the average of the remaining two tests plotted. The damage reduction of the second column of Table VII is not explicit, but is based on the above assumption.

d. Bolt Preload Variation

An evaluation of bolt preload, measured as nut tightening torque, was necessary in the course of determining damage reduction factors due to reaming. A .06 inch ream in a Loaded Hole or Joint specimen necessitates the installation of larger bolts. Testing with these bolts produced fatigue lives in the order of three times the basic values. It thus became necessary to test specimens with the larger bolts without the reaming process in order to evaluate the effect of reaming as such.

TABLE V

DAMAGE REDUCTION FACTORS FOR .03 INCH REAM AND .06 INCH REAM

LOADED HOLE SPECIMENS

1	N-FAILURE	122500	33900	16320	20500	347000	100000	33000	18700
2	N-REAM	48400	13900	4800	13900	48400	13900	4800	13900
3	REAM, IN.	.03	.03	.03	.03	.06	.06	.06	.06
4	TEST NO.	22(b)	22(c)	22(d)	39(c)	23(b)	23(c)	23(d)	39(d)
5	MAXIMUM STRESS	29000	33800	38500	33400	28700	33400	38300	33400
6	BASIC LIFE	106000	26500	9500	30000	120000	30000	9800	30000
7	TEST NO. (FIG. 59)	11	11	11	11	11	11	11	11
8	FINAL TORQUE	35	35	35	0	70	70	70	0
9	BASIC LIFE (TORQUE)	106000	26500	9500	20400	--	79500	--	20400
10	TEST NO. (FIG. 59 , TABLE XLI)	11	11	11	42(a)	--	42(d)	--	42(a)*
11	LIFE RATIO $\boxed{1}$ / $\boxed{6}$	1.155	1.280	1.718	.683	1.890	3.360	3.370	.622
12	AVERAGE LIFE RATIO	→	→	1.384	--	→	→	3.210	--
13	N-NET $\boxed{1}$ - $\boxed{2}$	74100	20000	11520	6600	298600	86100	28200	4800
14	NET LIFE RATIO $\boxed{13}$ / $\boxed{9}$	.700	.754	1.213	.324	--	1.085	--	.236
15	1.000 - $\boxed{14}$	.300	.246	--	.676	--	--	--	.764
16	REAM DAMAGE $\boxed{2}$ / $\boxed{6}$	.457	.524	.505	.463	.403	.463	.490	.463
17	DAMAGE REDUCTION $\boxed{16}$ - $\boxed{15}$	.157	.278	.505	--	--	.463	--	--
18	DAMAGE REDUCTION RATIO $\boxed{17}$ / $\boxed{16}$	.344	.531	1.000	--	--	1.000	--	--
19	AVERAGE DAMAGE REDUCTION	--	--	.625	--	--	--	--	--

* .250 IN. DIA. LIFE USED. .312 NOT AVAILABLE.

Reference Table VI for general notes.

TABLE VI

DAMAGE REDUCTION FACTORS FOR .03 INCH REAM.

4 BOLT JOINT SPECIMENS

1	N-FAILURE	55900	20600	5980	25200	12100	24900
2	N-REAM	31900	9800	4000	9800	9800	9800
3	REAM, IN.	.03	.03	.03	.03	.03	.03
4	TEST NO.	24(b)	24(c)	24(d)	31(c)	41(c)	33(c)
5	MAXIMUM STRESS	27700	31600	38100	31400	31400	31300
6	BASIC LIFE	60000	20000	7300	21000	21000	22000
7	TEST NO. (FIG. B33)	13	13	13	13	13	13
8	FINAL TORQUE	35	35	35	12	0	35
9	BASIC LIFE (TORQUE)	60000	20000	7300	18500	15300	22000
10	TEST NO. (FIG. B33 TABLE B44)	13	13	13	FIG.17	43(a)	13
11	LIFE RATIO $\boxed{1}/\boxed{6}$	.931	.959	.819	1.201	.576	1.133
12	AVERAGE LIFE RATIO	⟶		.903	--	--	--
13	N-NET $\boxed{1}-\boxed{2}$	24000	10800	1980	15400	2300	15100
14	NET LIFE RATIO $\boxed{13}/\boxed{9}$	.400	.540	.271	.832	.150	.685
15	1.000 - $\boxed{14}$	.600	.460	.729	.168	.850	.315
16	REAM DAMAGE $\boxed{2}/\boxed{6}$	.532	.490	.548	.467	.467	.446
17	DAMAGE REDUCTION $\boxed{16}-\boxed{15}$	0	.030	0	.299	0	.131
18	DAMAGE REDUCTION RATIO $\boxed{17}/\boxed{16}$	0	.061	0	.640	0	.294
19	AVERAGE DAMAGE REDUCTION	⟵ NEGLIGIBLE ⟶					

NOTES:

(1) Values in rows 1, 2, 3, 5, are taken from the data sheets for the tests referenced in row 4. Values for row 6 are taken from the curves referenced in row 7. Values for row 9 are taken from the curves and tables referenced in row 10.

(2) Ream damage, row 16, $D_0 = n/N$, where n is ream cycles, and N is basic life.

(3) Damage reduction ratio, row 18, is damage reduction factor, the ratio of ΔD, row 17, to D_0, row 16.

TABLE VII

DAMAGE REDUCTION FACTORS FOR .06 INCH REAM.

4 BOLT JOINT SPECIMENS

1	N-FAILURE	208000	114000	43500	66600	34700
2	N-REAM	31900	9800	4000	9800	9800
3	REAM, IN.	.06	.06	.06	.06	.06
4	TEST NO.	25(b)	25(c)	25(d)	33(b)	33(d)
5	MAXIMUM STRESS	27600	31400	38000	31300	31300
6	BASIC LIFE	61000	21000	7500	22000	22000
7	TEST NO. (FIG. 64)	13	13	13	13	13
8	FINAL TORQUE	70	70	70	45	70
9	BASIC LIFE (TORQUE)	--	102700	--	43400	20000
10	TEST NO.(TABLES LII, LIV)	--	ASSUME	--	43(c)	45(d)
11	LIFE RATIO $\boxed{1}$ / $\boxed{6}$	3.410	5.430	5.80	3.030	1.580
12	AVERAGE LIFE RATIO		→	4.88	--	--
13	N-NET $\boxed{1}$ - $\boxed{2}$	176100	104200	39500	56800	24900
14	NET LIFE RATIO $\boxed{13}$ / $\boxed{9}$	--	1.016	--	1.310	1.245
15	1.000 - $\boxed{14}$	--	--	--	--	--
16	REAM DAMAGE $\boxed{2}$ / $\boxed{6}$	.524	.467	.534	.445	.445
17	DAMAGE REDUCTION $\boxed{16}$ - $\boxed{15}$	--	--	--	--	--
18	DAMAGE REDUCTION RATIO $\boxed{17}$ / $\boxed{16}$	--	1.000	--	1.000	1.000
19	AVERAGE DAMAGE REDUCTION	--	--	--	--	1.000

Reference Table VI for general notes.

The results of the bolt preload evaluation are plotted in Figures 15 through 18. The basic variation of fatigue life as a function of bolt torque for Loaded Hole specimens is shown as the solid line of Figure 15. The torque values from zero to 35 inch-pounds were taken from specimens with 1/4 inch diameter bolts installed, consistent with the original specimen design. A bolt torque of 35 inch-pounds is the standard value used for shear applications. The data for 45 inch-pounds and 70 inch-pounds were from specimens with the larger 5/16 inch diameter bolts installed, with 70 inch-pounds being the standard value for this attachment size (Reference 13).

The dashed line, Figure 15, is for comparable test results with peened specimens. The one point which indicates a significant increase in fatigue life is the result of specimens tested with 1/4 inch diameter bolts installed without measured torque. Two sets of specimens were tested with graphite grease (anti-seize compound) applied between the doubler plates and the specimens as a variation of friction between the plates. In similar tests of other specimens, the greased specimen test results were unchanged from the dry. It is assumed, therefore, that the fatigue life shown for 25 inch-pounds torque is due to a higher bolt preload as a result of greased threads as compared to dry threads.

The effect of reaming Loaded Hole specimens at 13,500 cycles is included in the fatigue life versus bolt torque plot of Figure 16.

The plots of results of 4 Bolt Joint specimens, Figure 17, are similar to those for the Loaded Hole specimens. These specimens are subject to high bearing stresses and have mating clad surfaces which become pitted when subjected to fretting (Figure 10). The increase in fatigue life once the standard 35 inch-pounds torque for the 1/4 inch diameter bolt was exceeded, was so spectacular that it was not feasible to fair the curve in that area. Hence, that effect is shown as a discontinuity of slope at the 35 inch-pounds torque value. 5/16 inch diameter bolts were installed for tests with torque values greater than 35 inch-pounds, except for the peened specimens with the torque not measured, and as noted for zero torque data.

The effect of reaming the 4 Bolt Joint specimens at 9,800 cycles is shown in Figure 18. The solid line represents the results of using a .03 inch ream up to 35 inch-pounds torque, and a .06 inch ream beyond that value. Two individual points are plotted for the results of a .06 inch ream at abnormally low torque values. Two points are plotted for the results of specimen tests which included peening after the reaming at 9,800 cycles.

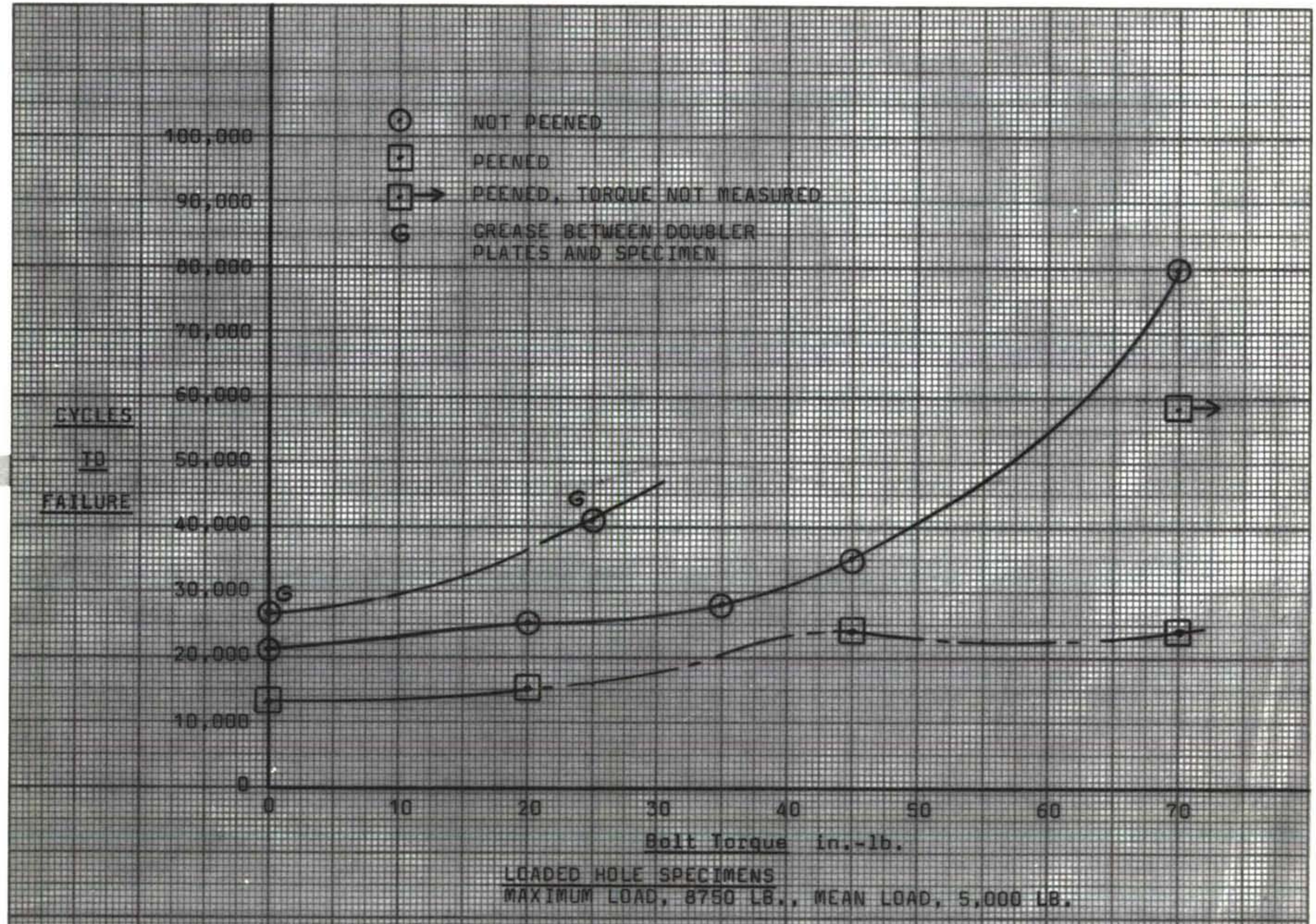


Figure 15. Loaded Hole Specimen, Fatigue Life versus Bolt Torque.

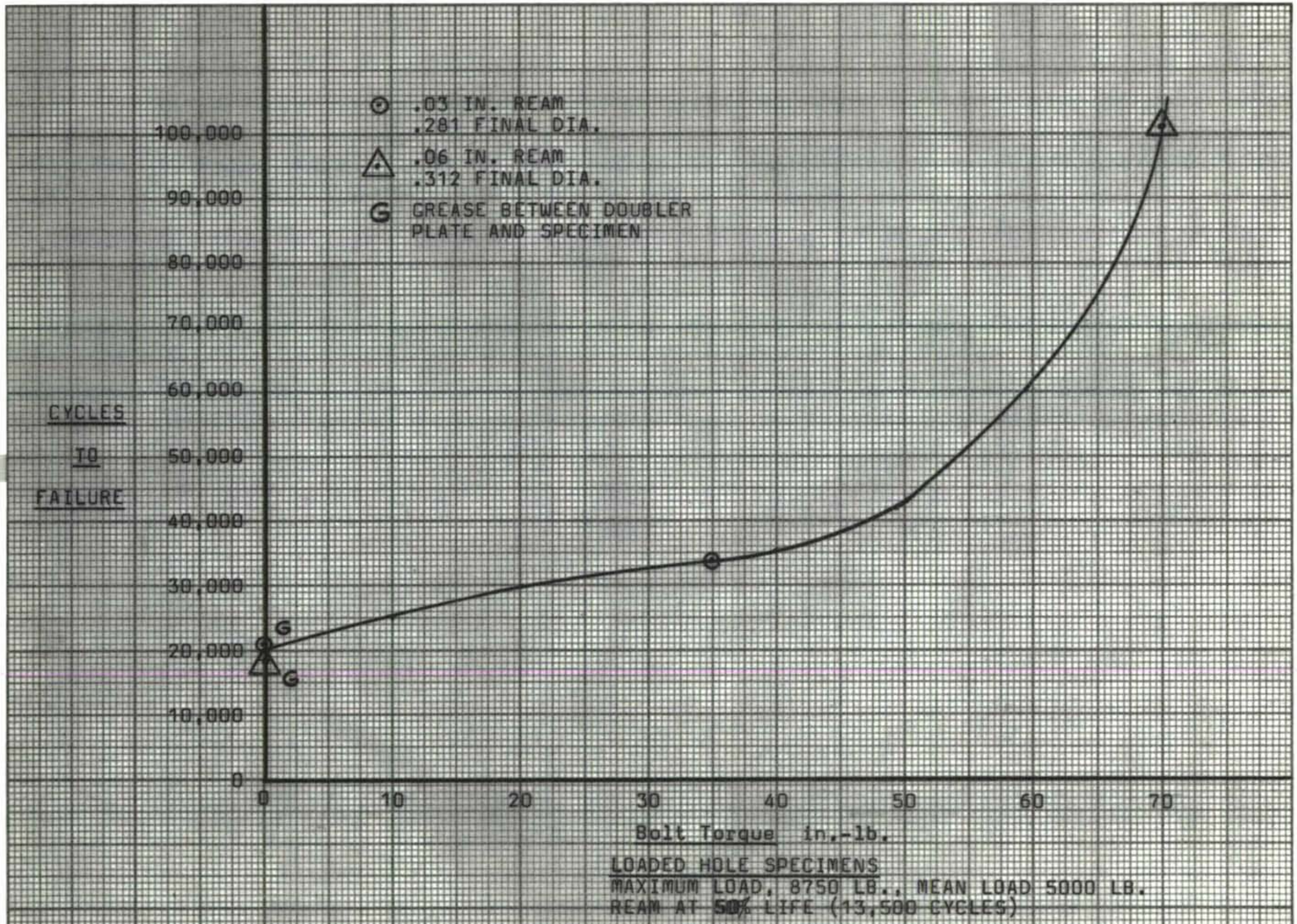


Figure 16. Loaded Hole Specimen Reamed at 50 Per Cent Basic Life, Fatigue Life versus Bolt Torque.

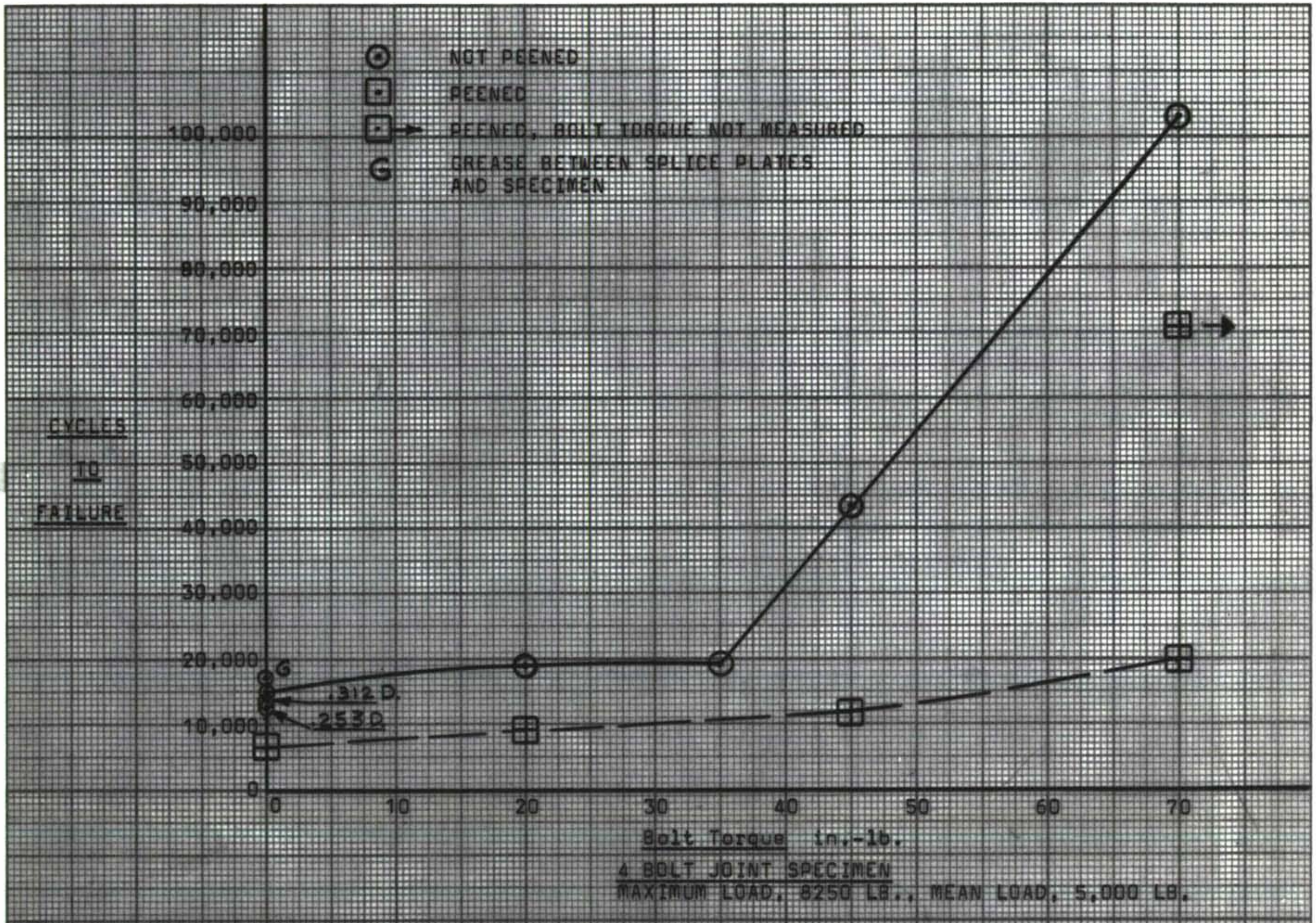


Figure 17. 4 Bolt Joint Specimen, Fatigue Life versus Bolt Torque.

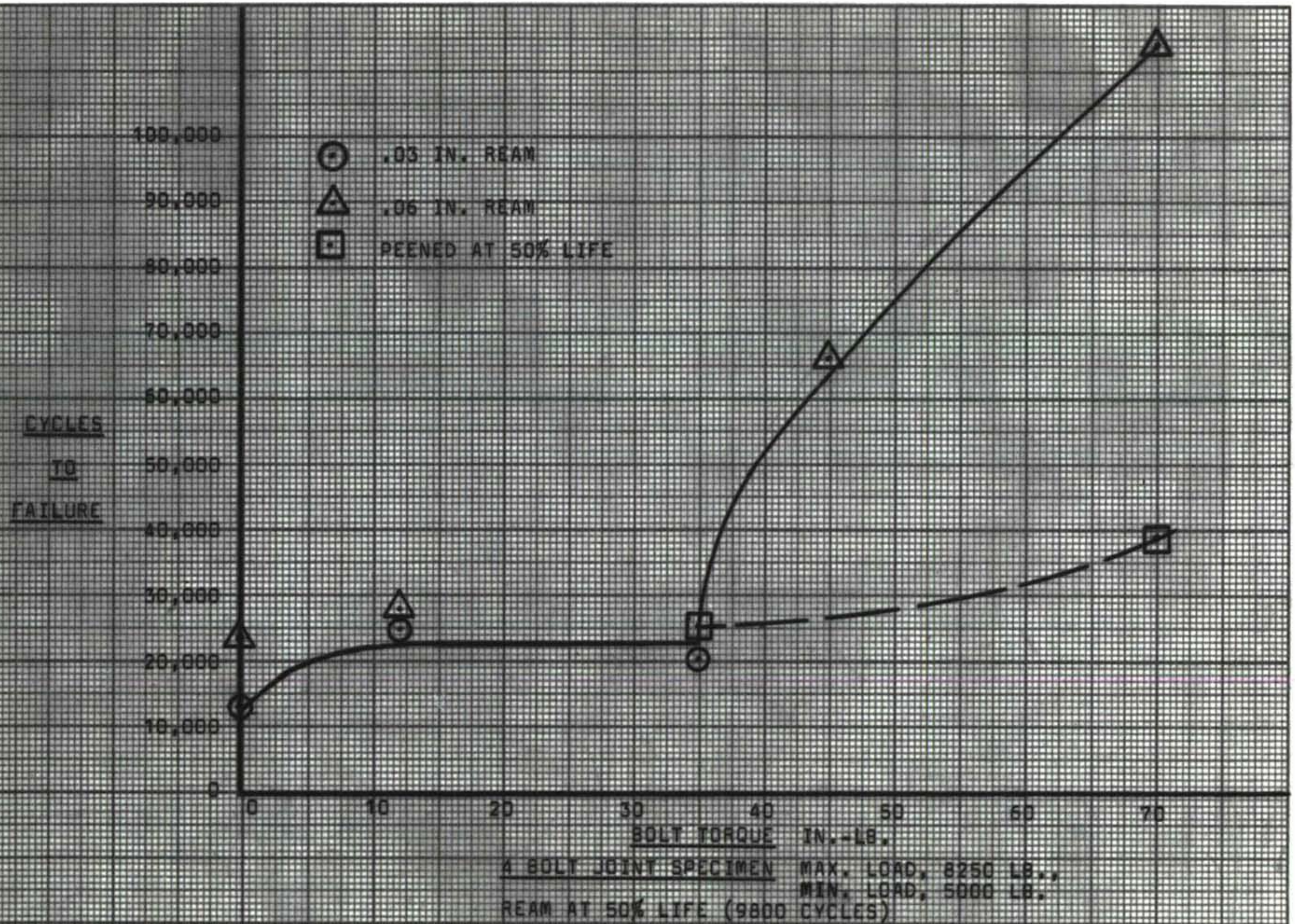


Figure 18. 4 Bolt Joint Specimen Reamed at 50 Per Cent Basic Life, Fatigue Life versus Bolt Torque.

Fatigue life versus bolt torque data are plotted in a more generalized form in Figure 19. Fatigue life data from the different specimen tests were adjusted to be consistent with a maximum stress of 34,000 psi, and maintaining a mean stress of 20,000 psi. A dimensionless bolt preload C, was derived as an expression of the ratio between preload and shear loads transferred by the bolt.

The dimensionless torque variable, C, is a ratio of a factor which is proportional to bolt preload, T/D, to the ultimate tensile strength of the part, or specimen, divided by the number of bolts, $F_{tu} \times A_{net}/B$.

$$C = \frac{T\ B}{D\ F_{tu}\ A_{net}}$$

where: T is nut tightening torque, inch-pounds,

B is number of bolts carrying the applied load,

D is nominal bolt diameter, inches,

F_{tu} is ultimate failure stress for the material,

A_{net} is the minimum net area subject to the applied tension loading.

The basic plots (Figure 19), of Loaded Hole and 4 Bolt Joint specimen results are taken from Figures 15 and 17, and are well defined. The results for peened specimens are defined to a limited extent. These curves were extended by assuming that not measured bolt torque values were 105 inch-pounds. Two individual points were available from 8 Bolt Joint tests. One of these points plots coincidental with the Loaded Hole curve with no necessary relationship existing between this point and the curve. Assumed torque values for the 8 Bolt Joint specimens are 17 inch-pounds for 3/16 inch diameter bolts, and 52 inch-pounds for the 1/4 inch diameter bolts.

Two parameters are believed to be significant in locating the various curves on a plot of fatigue life versus C for constant maximum and mean stresses. The first is the bolt hole bearing stress as compared to the net tension stress. The higher the bearing stress, the lower the fatigue life for C = 0. Secondly, the combination of materials used as primary structure and as splice or doubler plates, including peened or not peened surfaces. Further investigation is necessary to define these curves for ranges of bearing stress ratios and material combinations.

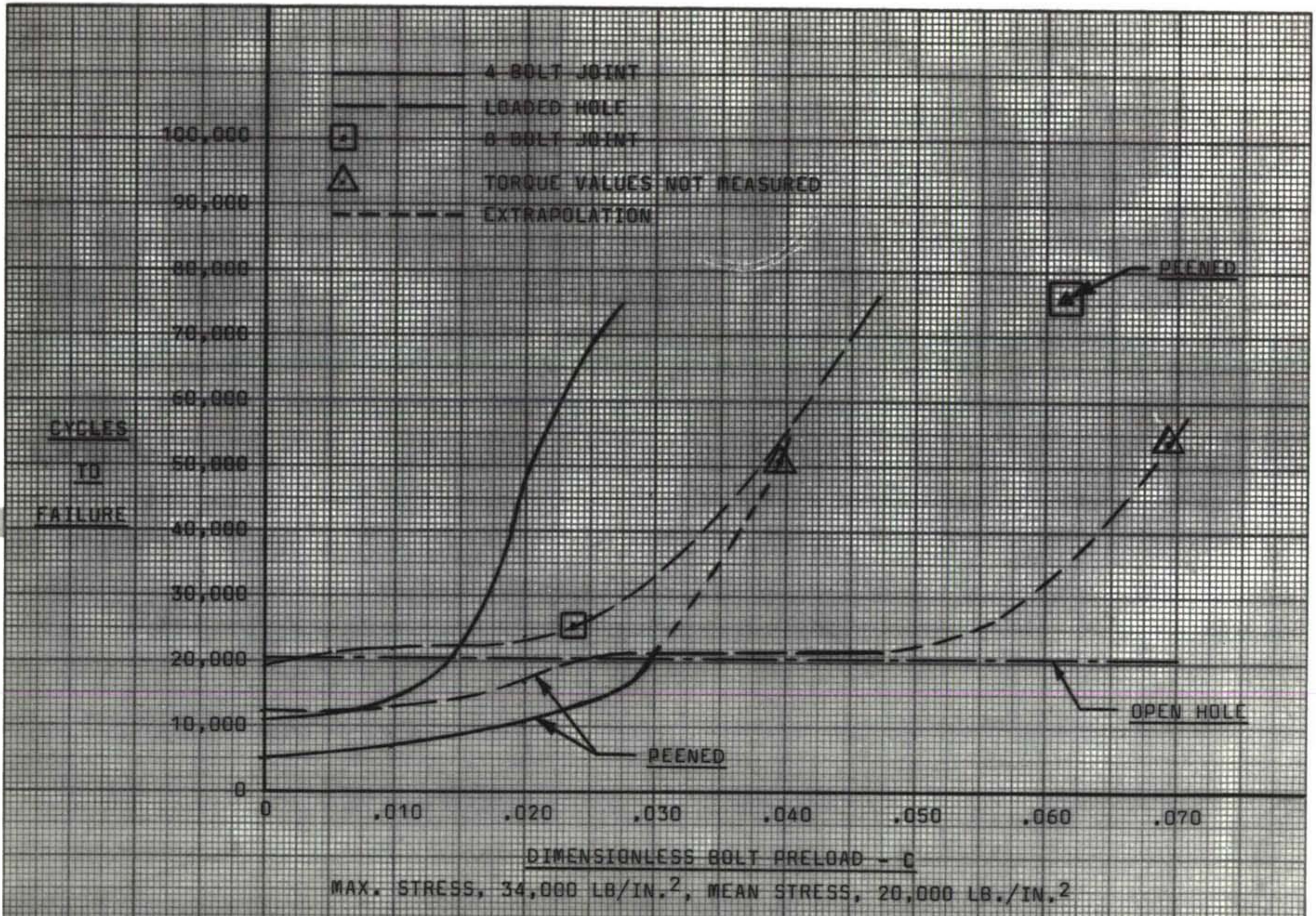


Figure 19. Fatigue Life versus Nondimensional Bolt Torque Factor.

e. Mixed Cycle Testing

Mixed loading cycles were applied to one set of 8 Bolt Joint specimens. Different combinations of loading cycles were applied to each group of four specimens, with the applied stresses and cycles to failure as shown in Table LVIII.

The linear cumulative damage rule (Palmgren-Miner Method) was used for computing damage. Loading cycles were changed after every increment of approximately .25 damage accumulation up to a total of 1.50. At .50 damage accumulation, .06 inch reaming and peening were applied. Once 1.50 damage had been achieved, loading cycles were held constant at a relatively high level until failure.

Average damage values at failure ranged from 4.92 to 6.77, Table VIII. A parallel computation is given with the assumptions of 100 per cent damage reduction (zero-time) due to reaming, and a factor of 4.0 applied to the fatigue life data used after reaming to account for the increase in fatigue life due to increased bolt preload. In all cases the damage computed with these factors included exceeded 1.00. Two groups of specimens achieved slightly higher damage values at failure than the remaining two, which indicates a tendency of the lower stress levels applied prior to the final loading to be more beneficial to life than the higher load cycles. These results correlate well with those plotted in Figure 15 of Reference (3).

The comparison between the results of mixed cycle testing and constant load testing is shown by the last two bars of the chart in Figure 12, with the specimens subjected to mixed cycle loading producing higher damage values at failure than those subjected to constant loading.

f. Edge Distance Limitations

No definite limitations were encountered as a result of small edge distance. As applied to the specimens tested in this program, edge distance is measured normal to the direction of applied load, and is expressed as the ratio of the distance from the center of the hole to the edge of the part to the hole diameter. Control curves for Open Hole specimens with edge distance ratios of 2.5, 2.0, and 1.5 are shown in Figures 38, 39, and 40, while the control for a symmetrically located hole is shown in Figure 32. The computed maximum stresses at the edge of a hole given as a function of applied load, P, are presented in Section III, Part 1.

TABLE VIII

CYCLE RATIO SUMMATION, MIXED LOADING CYCLES.

8 BOLT JOINT SPECIMENS

Mean Stress	Maximum Stress	Cycles Per Block, n	Cycle Ratio Summation N	Cycle Ratio Summation n/N	Cycle Ratio Summation $\sum n/N$	Cycle Ratio Summation, Zero-Time, 4N: 4N	Cycle Ratio Summation, Zero-Time, 4N: n/N	Cycle Ratio Summation, Zero-Time, 4N: $\sum n/N$
19,100	31,400	15,400	57,000	.270	.27	--	0	--
19,100	26,800	38,200	270,000	.142	.41	--	0	--
9,550	25,000	24,900	94,000	.265	.68	376,000	.066	.07
9,550	19,100	59,300	270,000	.220	.90	1,080,000	.055	.13
9,550	22,300	40,700	148,000	.274	1.17	592,000	.069	.20
9,550	31,400	207,500	37,000	5.60	6.77	148,000	1.403	1.60
9,550	25,000	24,900	94,000	.265	.27	--	0	--
9,550	19,100	59,300	270,000	.220	.49	--	0	--
19,100	31,400	15,400	57,000	.270	.76	228,000	.068	.07
19,100	26,800	38,200	270,000	.142	.90	1,080,000	.035	.11
19,100	27,700	48,000	175,000	.274	1.17	700,000	.069	.18
19,100	38,200	68,200	16,000	4.26	5.43	64,000	1.068	1.25
19,100	38,200	3,800	16,000	.238	.24	--	0	--
9,550	19,100	59,300	270,000	.220	.46	--	0	--
19,100	38,200	3,800	16,000	.238	.70	64,000	.060	.06
9,550	19,100	59,300	270,000	.220	.92	1,080,000	.055	.12
19,100	38,200	3,800	16,000	.238	1.16	64,000	.060	.18
9,500	19,100	59,300	270,000	.220	1.38	1,080,000	.055	.24
19,100	38,200	56,700	16,000	3.54	4.92	64,000	.885	1.13
9,550	19,100	59,300	270,000	.220	.22	--	0	--
19,100	38,200	3,800	16,000	.238	.46	--	0	--
9,550	19,100	59,300	270,000	.220	.68	1,080,000	.055	.06
19,100	38,200	3,800	16,000	.238	.92	64,000	.060	.12
9,550	19,100	59,300	270,000	.220	1.14	1,080,000	.055	.18
19,100	38,200	84,500	16,000	5.28	6.42	64,000	1.322	1.50

Double reaming was applied to the specimens of test numbers 40 and 50, with the results as given in Tables XXVIII and XXIX. Edge distance ratios before reaming were 2.5 and 1.5; a .06 inch ream was applied after the first block of load cycles, and a .03 inch ream after the second block. These specimens proved to have lower basic fatigue lives than those of the control curves, with the effective damage accumulation much higher than the scheduled 50 per cent basic life at the time of the first ream. The relatively low cycles to failure values produced by these tests are attributed to the particular specimens. No particular limitation due to edge distance was evident.

All Loaded Hole and Joint specimens were designed with symmetrical hole patterns. The 8 Bolt Joint specimens, after the holes were reamed to .250 inch diameter, had an edge distance ratio of 1.5. The results of the 8 Bolt Joint tests compare favorably with those of the Loaded Hole and 4 Bolt Joint specimens.

The ratio of bolt spacing parallel to the direction of applied load to bolt diameter was most critical for the 4 Bolt Joint specimen with holes reamed to .312 inch diameter. This ratio was 3.2 as compared with a general industry standard of a minimum of 4.0. This effect did not produce any limitation upon fatigue life. (Reference the bar chart of Figure 12 for the relationship between the results of the reamed joint specimens.)

The effects of edge distance and bolt spacing are factors which must be considered in any given structural analysis. However, it is concluded from the results of this testing program that no abnormal limitations are applicable due to bolt location geometry.

g. Conclusions

It is recommended that the .03 inch reaming process be avoided wherever possible. The effect on fatigue life is favorable for most applications, but it is definitely marginal for highly loaded joints. It has the added disadvantages of requiring special bolts after reaming without any benefit of increased preload, and of not cleaning up holes which are more than .015 inches beyond the nominal radius, such as egg shaped or out-of-round holes. During the A-26A permanent repair installation, this latter condition occurred, and influenced the decision to take all critical holes out to the next nominal diameter.

The .06 inch ream is strongly recommended for increasing the fatigue life of existing structures. Damage reduction is reliable and the installation of a larger fastener will produce a longer basic fatigue life. In the event that fatigue life versus C curves become defined for a practical range of structures, bolt sizes can be selected with confidence for the purpose of producing significantly longer fatigue lives.

8. Peening

a. General

The basic peening process used in the specimen testing program was identical to that used on the production A-26A airplane. This process was developed during the A-26A wing cyclic fatigue test program for the purpose of peening in and around small screw holes. A deflector was designed for directing shot to the inside surface of screw holes with diameters as small as .188 inch (Figure 20). This process was applied to the critical areas of the wing spar caps of the A-26A service airplanes as one of the requirements of the Permanent Repair modification developed during the fatigue test program.

Although shot peening was used for processing the cyclic test wings, glass peening was used on the service aircraft in order to avoid corrosion problems, and was used in the basic process for the specimen testing. The peening intensity attained on a surface inside of a screw hole, using the deflector of the basic process, .020 inch diameter glass, is estimated to be .007A.

The objective of the peening process is to inhibit the development of fatigue cracks by producing residual compression stresses on the surface material of a part in areas where fatigue failures are likely to occur. These residual stresses are produced by deforming, or cold working, the surface material due to the impact of the shot blasting. Figure 11 of Reference (11) indicates that creating a compressive residual stress to a depth of .008 inch under the surface produces a maximum stress of 50,000 psi on 7075-T6 aluminum alloy. This was achieved using an Almen intensity of .007A (.002C).

Reference (2) gives values for residual stresses in 2014-T6 due to peening with an intensity of .009A. Maximum compressive stress is 39,000 psi, tapering off to 5,000 psi at .030 inch under the surface. Test specimens are made from 2014-T6 and peened with a basic intensity of .007A.

b. Open Hole Peening

The basic peened Open Hole test results did not show any significant improvement in fatigue life as compared to the basic not-peened test results. An additional series of peened specimen tests was scheduled in order to find the best process for extending fatigue life:

(a) Test number 35. The clad material was removed from the surface area around the hole prior to peening.

(b) Test number 37. The basic peening process was applied to a .250 inch diameter hole as compared with the .188 inch basic diameter.

(c) Test number 30. A peening intensity of Almen .015A was used without using the deflector for peening the inside hole surface. The inside hole surface was peened by directing the shot obliquely from the outside. The hole diameter was .250 inch, and the surfaces clad.

(d) Test number 29 was the same as test number 30 except that the clad material was removed from the surface area around the hole.

(e) Test number 36 was the same as test number 30 except that the specimen thickness was .125 inch as compared to the basic .250 inch.

Peened surfaces are shown in Figure 21, magnified 200 times. Figure 21 (a) shows the inside surface of a hole peened with the basic deflector process. This specimen was soaked at 940°F for 30 minutes followed by a cold water quench. The cold worked area is shown by the smaller grain sizes. Depth of cold working is estimated at .0015 inch.

Figure 21 (b) shows an outside surface peened to an intensity of .015A. This specimen was not heat treated. Estimated cold working depth is approximately the same as that found on the inside hole surface peened at .007A.

During the course of analyzing test results, it was noted that each group of data taken from peened specimen results was remarkably consistent. A survey of the sample standard deviations, s, indicated that the deviation values (scatter) were consistently smaller for peened specimen tests than for not-peened specimen tests. A typical compilation of test results is plotted on probability paper, Figure 22, showing graphically the relative scatter of peened and not-peened specimen test results. These data are all taken from the same nominal loading level, with the failure cycles adjusted to account for minor variations in maximum stresses.

c. <u>Loaded Hole and Joint Peening</u>

Peened Loaded Hole and Joint specimens generally had fatigue lives that were equal to, or somewhat less than those for not-peened specimens tested under comparable conditions.

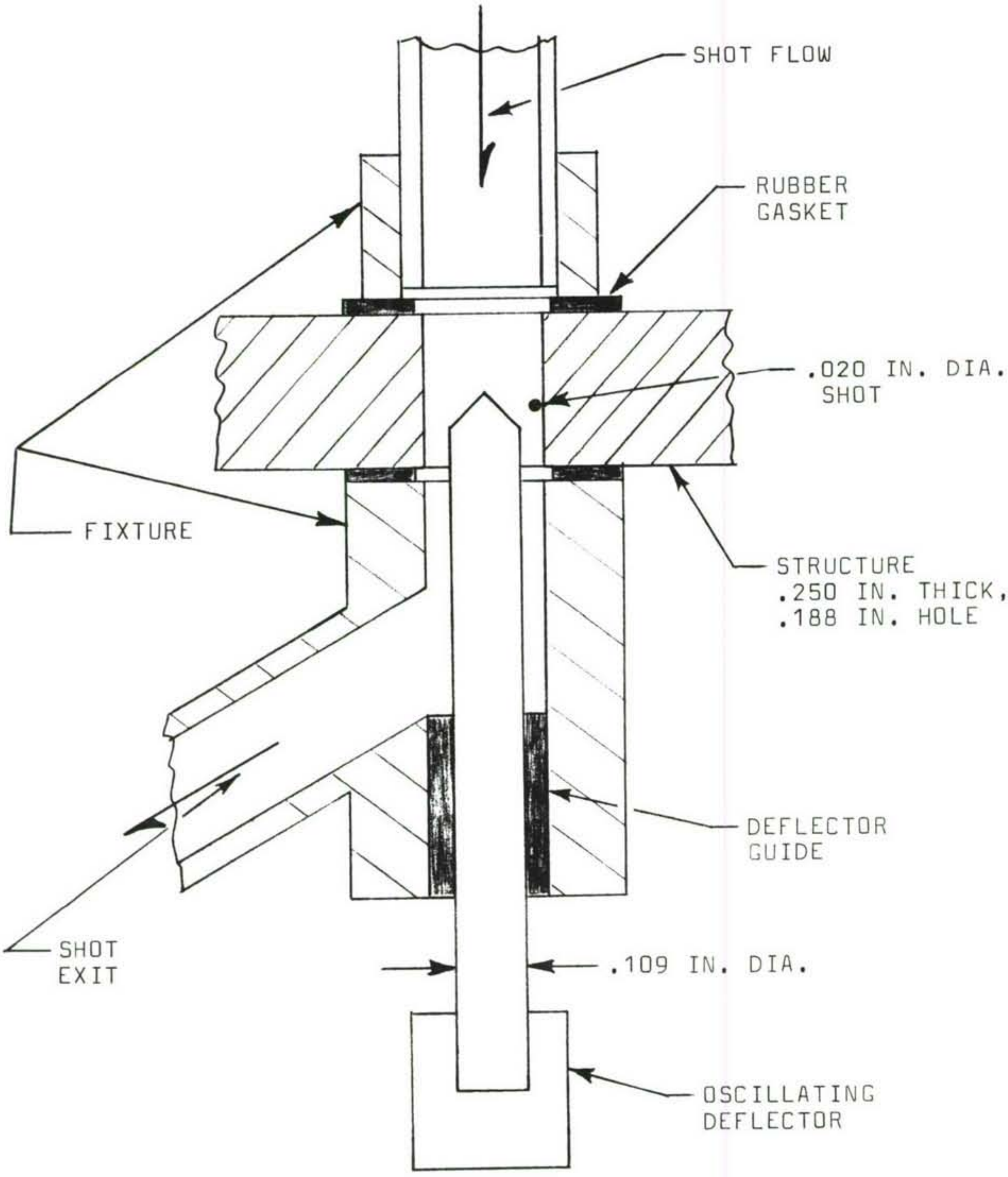


Figure 20. Sketch Showing Section View of Peening Device. Scale: 4:1

Figure 21 (a)

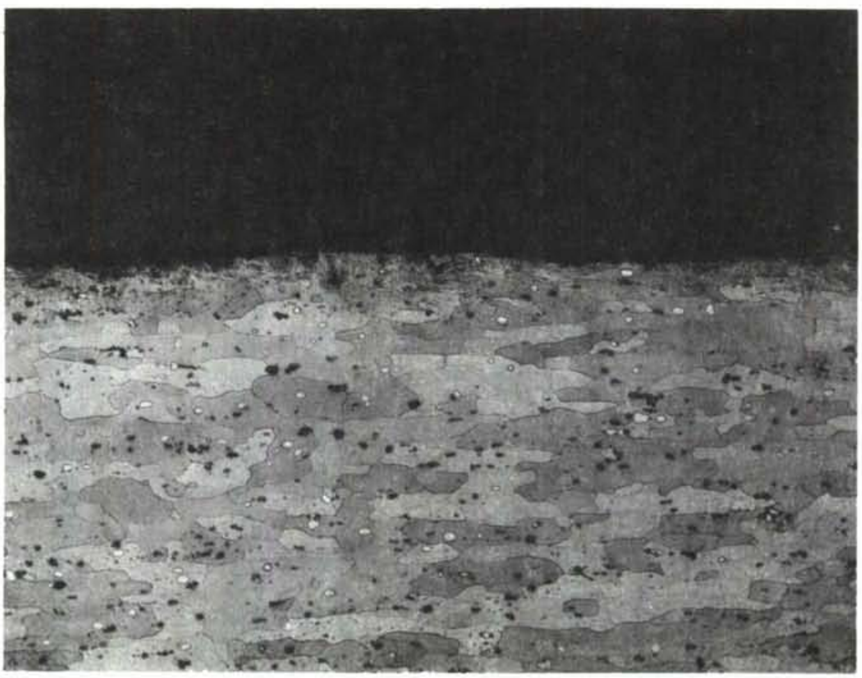


Figure 21 (b) (Concluded)

Figure 21. Peened surfaces magnified 200 times. (a) Inside of hole peened with deflector; (b) Outside surface.

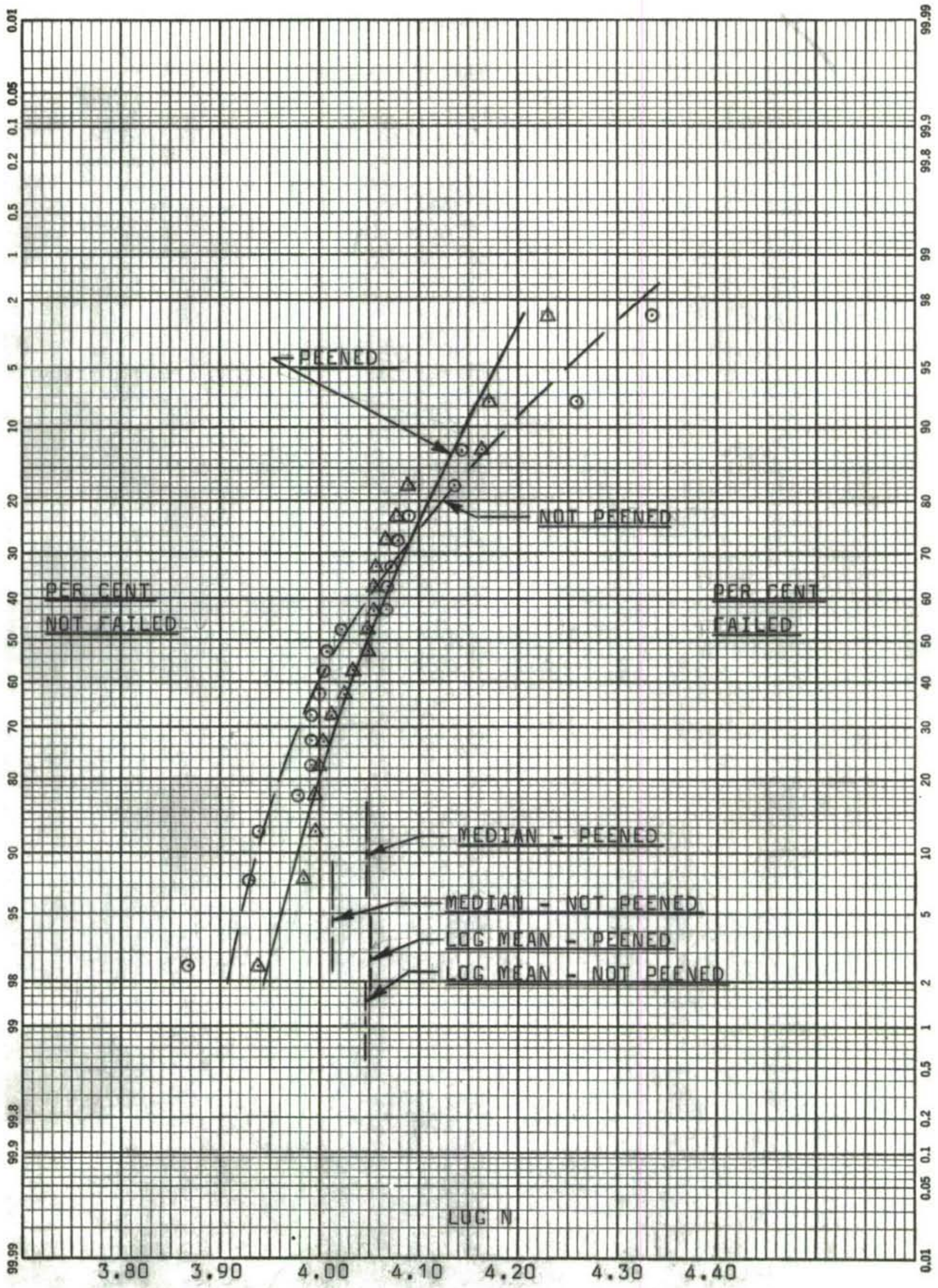


Figure 22. Per Cent Failure versus Log N For Peened and Not Peened Specimens, Maximum Stress, 38,000 lb/in^2.

Figures 15, 17, and 19 are plots of fatigue life versus bolt torque for constant loading levels and show a reduction of fatigue life for the peened specimens as compared to the not peened. It is concluded that there are values of bolt torque which bring the peened specimen test results up to those for the not peened specimens. In every case where comparable fatigue lives were achieved by peened specimens, bolt torque was not measured, leading to the conclusion that, from extrapolating the fatigue life versus bolt torque curves, the not measured torque was higher than any of the measured values.

Two sets of tests were run with peened specimens without measured bolt torque:

(a) Basic peened Loaded Hole and 4 Bolt Joint tests, numbers 27 and 28. These specimens were tested prior to the evaluation of the effect of bolt torque on fatigue life. The specimens were reassembled after the peening process had been applied using standard wrenches rather than torque measuring wrenches. The fatigue life results of these tests were scattered, with some values essentially the same as for not peened specimens, and some higher values indicative of the effect of higher bolt preloads.

(b) All of the 8 Bolt specimens that were reamed to .250 inch diameters (Test numbers 47 and 49) had bolts with not measured torque values. This was done as a matter of expediency, as the normal socket used with the torque wrench did not clear with the extremely close bolt spacing. This condition simulates that of service airplanes as torque wrenches are not normally required for skin to spar attachments.

It is concluded that for mating flat surfaces, peening produces an irregular surface for transmitting friction forces, and as such, produces highly localized stress variations. The interaction of one of these localized stress areas with the normal stress increase at the edge of a hole causes a decrease in fatigue life. For sufficiently high bolt preload values, the irregular surface is essentially flattened out and the stress due to friction becomes more uniform.

d. Conclusion

Based on the existing state-of-the-art peening in and around small screw holes in aluminum structural members, peening cannot be recommended as a means of increasing the fatigue life of aluminum parts with critical small screw holes.

SECTION IV

A-26A SERVICE LIFE PREDICTION

Damage accumulation versus service hours for the A-26A airplane is given in the A-26A Service Life Prediction, Reference (1). Damage accumulation for the three most critical sections of the wing are shown on pages 1-4, 1-5, and 1-6, of that report. These plots are based upon an analysis of the wing cyclic fatigue test results and measured A-26A airplane operational flight loads (Vgh) data.

One of the objectives of the specimen testing program is to correlate, or modify, the assumptions made in the analysis of the wing cyclic test results.

The analysis of the wing cyclic test results included the following:

(a) Evaluation of initial damage. The A-26A wings selected for the cyclic fatigue test had previous service time as a B-26B airplane of approximately 3500 hours. Initial existing damage was evaluated for critical stations from wing cyclic test results.

(b) Evaluation of damage reduction due to reaming. A damage reduction factor of .7 was used for both reaming processes, .03 inch and .06 inch reams.

(c) Evaluation of damage accumulation rates as a function of cyclic test service hours. This was accomplished by selecting K_t values originally calculated from component tests, unit stress calculations for given critical sections, and the fitting of plots to match 100 per cent damage for each of several failures, or cracks, that occurred during the cyclic test. A reduced K_t value used after the peening process was applied to account for the reduced damage accumulation rates which were assumed to be due to the effect of having peened screw holes.

The time-history of the damage accumulation for the most critical section of the A-26A wing is reproduced in Figure 24. The only net change to this plot caused by the conclusions drawn from this specimen testing program is due to the damage reduction factor for the .06 inch ream. This reaming process was applied as one of the requirements of the Permanent Repair, a modification applied to all A-26A airplanes as a result of the cyclic fatigue testing program.

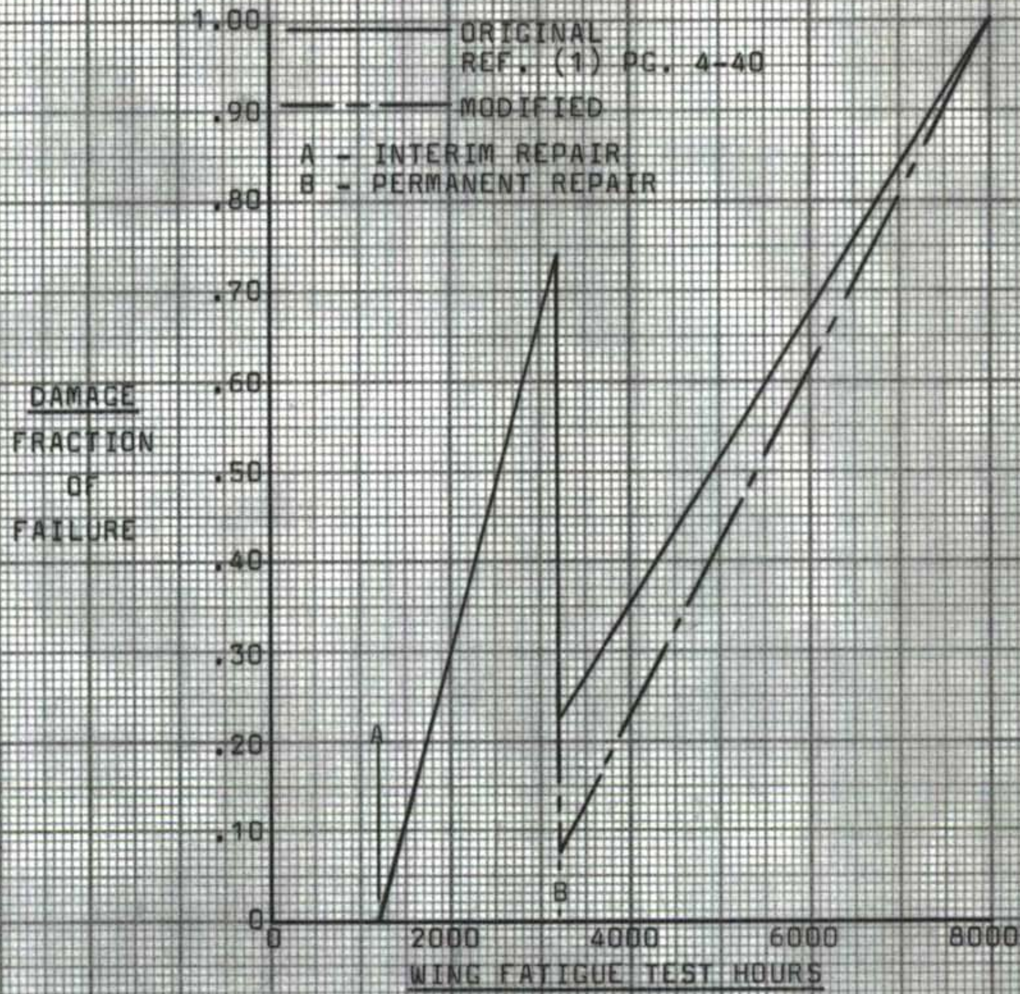


Figure 23. A-26A Cyclic Fatigue Test Damage Accumulation Time-History. Front Spar. Station 46.

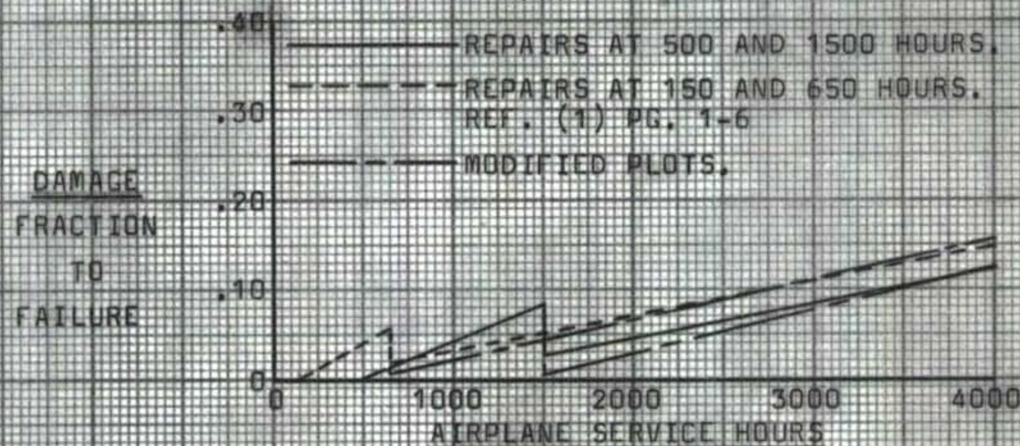


Figure 24. Damage Accumulation versus A-26A Airplane Service Hours. Front Spar. Station 46.

The damage reduction factor of .70 used for both of the reaming processes is unchanged for the .03 inch ream and is changed to .90 for the .06 inch ream. This change produces the modification shown for the cyclic test damage time-history of Figure 23.

The remaining cyclic test time-histories are unchanged. The reduction of damage accumulation rates after the installation of the Permanent Repair has now been attributed to the increased preloads associated with the larger bolts installed because of reaming critical holes, rather than to the effects of shot peening.

The damage accumulation rate reduction factor for the A-26A wing front spar, station 46, Figure 23, is approximately 2.0. Specimen results of .06 inch reaming tests indicate reduction rates of 2.5 to 3.0. Because the attachments at this wing station are subject to redundant loads from the steel reinforcement strap to the wing spar, it is assumed that the bearing loads on the attachments were increased after the change from 3/16 inch diameter bolts to 1/4 inch diameter during the Permanent Repair installation. Thus, the increased rigidity of the attachments produced increased loads, as compared to the specimen tests, with constant loading being applied before and after a reaming process.

The A-26A Service Life Prediction damage accumulation for the most critical wing station is shown in Figure 24. The unmodified damage plot of Reference (1), was calculated using the damage accumulation rates and damage reduction factors taken from the analysis of the cyclic test results, and airplane useage computed from Vgh data measured on service airplanes. The useage for the Service Life Prediction varies from that of the cyclic test loading because of the difference between measured data and estimated data, and because the Service Life Prediction contains no conservatism factors, while a factor of 2.0 was applied to the frequency of load application for the cyclic test. Conservatism factors, accounting for variations in airplane useage and for scatter in fatigue failure data, are applied to the Service Life Prediction damage accumulation by the airplane operators.

The Service Life Prediction damage accumulation of Figure 24 is modified by applying ratios taken from the changes in damage reduction and damage accumulation rate of the cyclic test time-history, Figure 23. These changes are considered to be minor, with a damage value of .157 at 4,000 service hours as compared with the original value of .149.

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13. Air Force Systems Command Manual: Handbook of Instructions for Aircraft Designers, Part B, Chapter 4.

APPENDIX I

DATA SUMMARY

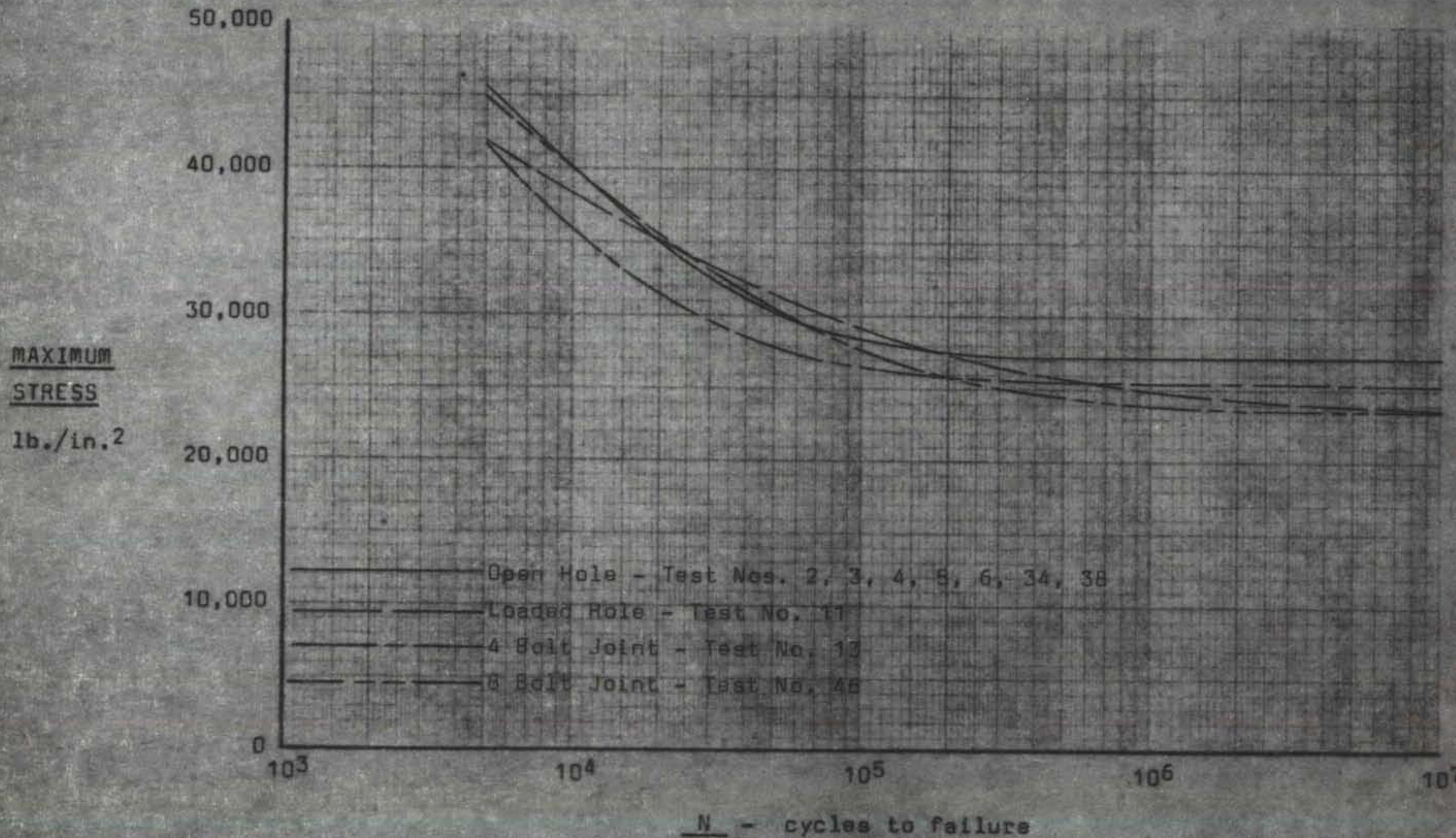

Figure 25. S-N Curve Comparison for Open Holes, Loaded Holes, 4 Bolt Joints, and 8 Bolt Joints. Mean Stress, 20,000 psi.

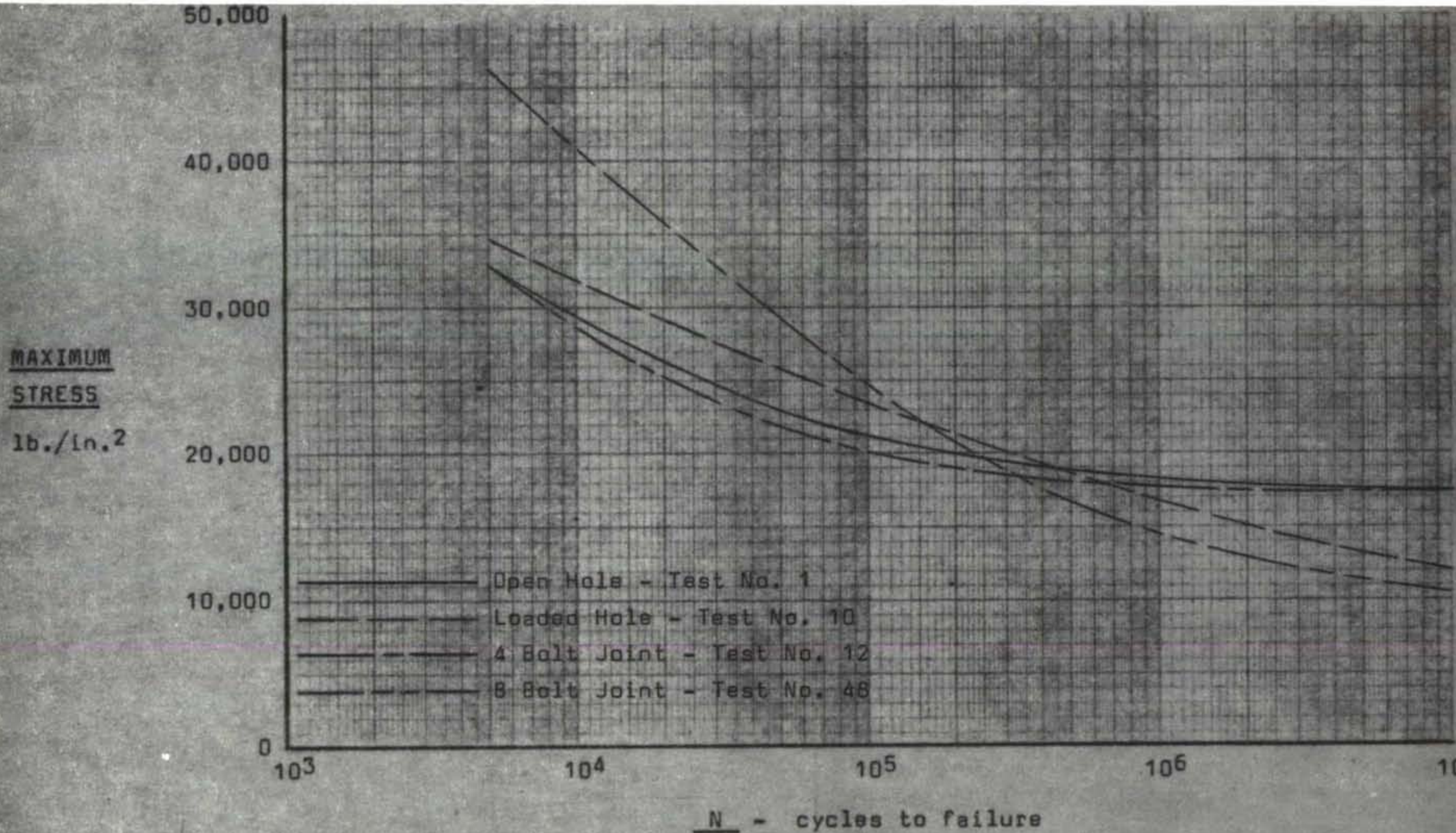

Figure 26. S-N Curve Comparison for Open Holes, Loaded Holes, 4 Bolt Joints, and 8 Bolt Joints. Mean Stress, 10,000 psi.

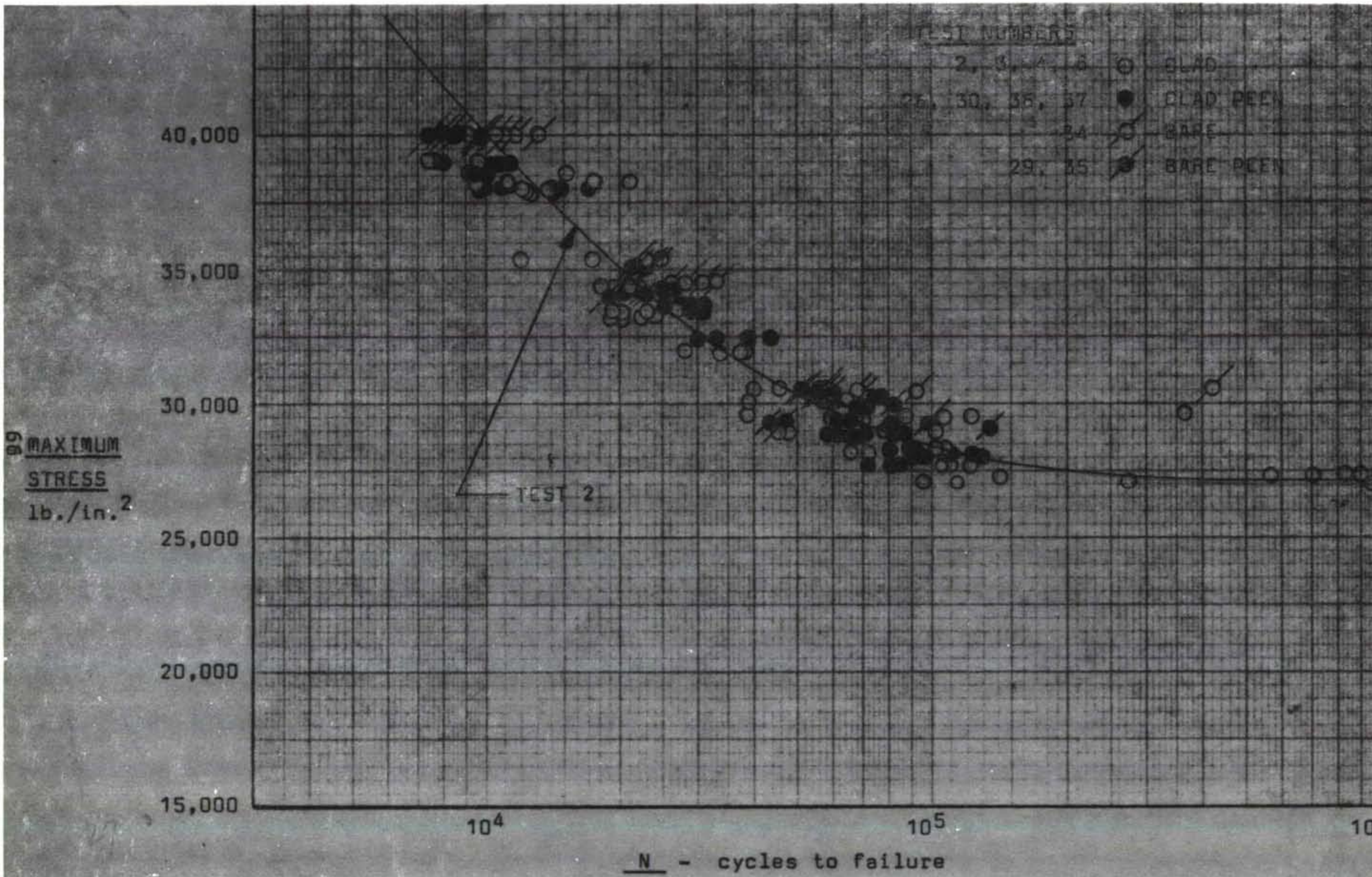

Figure 27. Specimen Test Results for Open Holes. Peened and Not Peened. 20,000 psi Mean Stress.

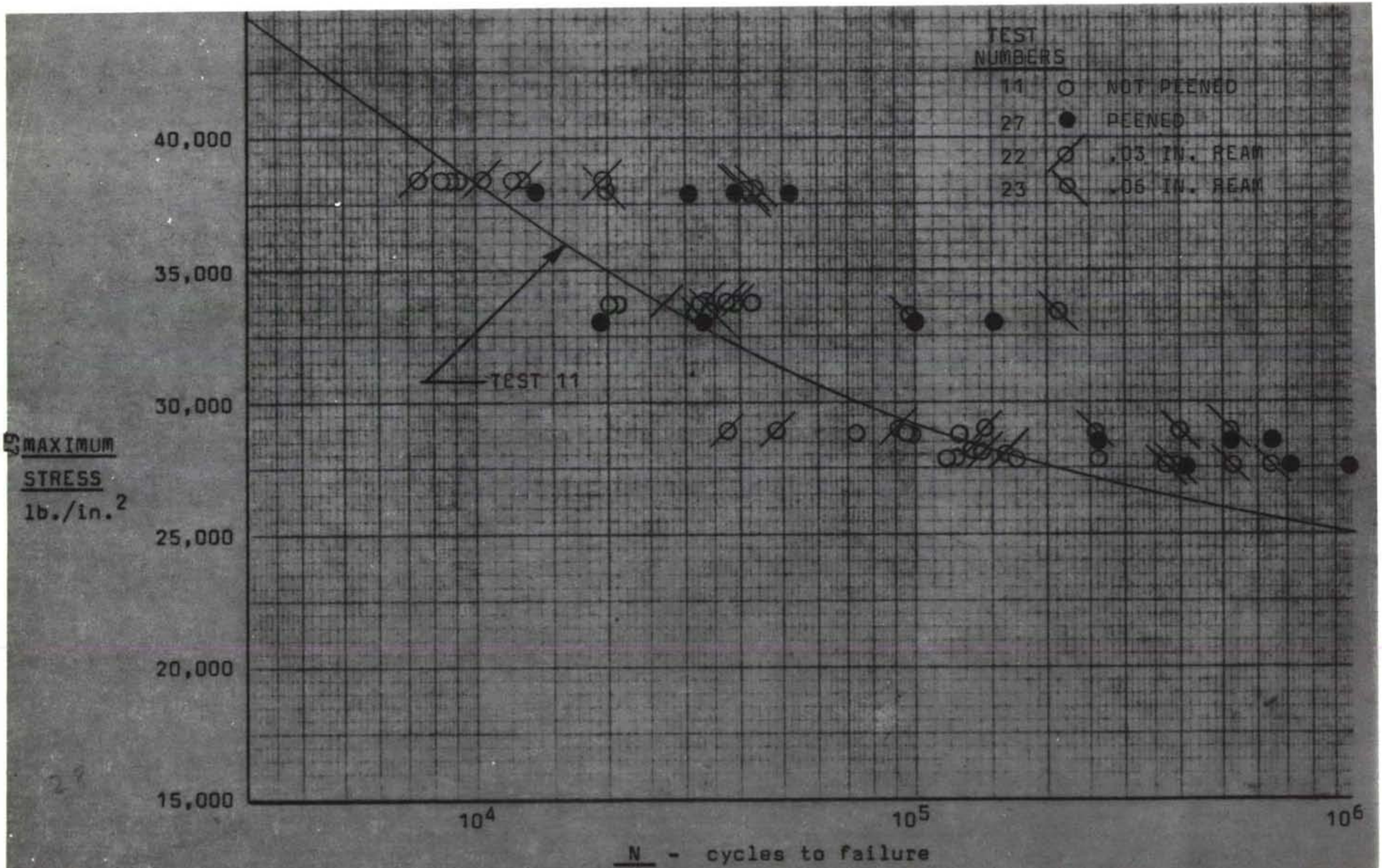

Figure 28. Loaded Hole Specimen Test Results. Not Peened, Peened (without Torque Control), .03 in. and .06 in. Ream at 50% Life. 20,000 psi Mean Stress.

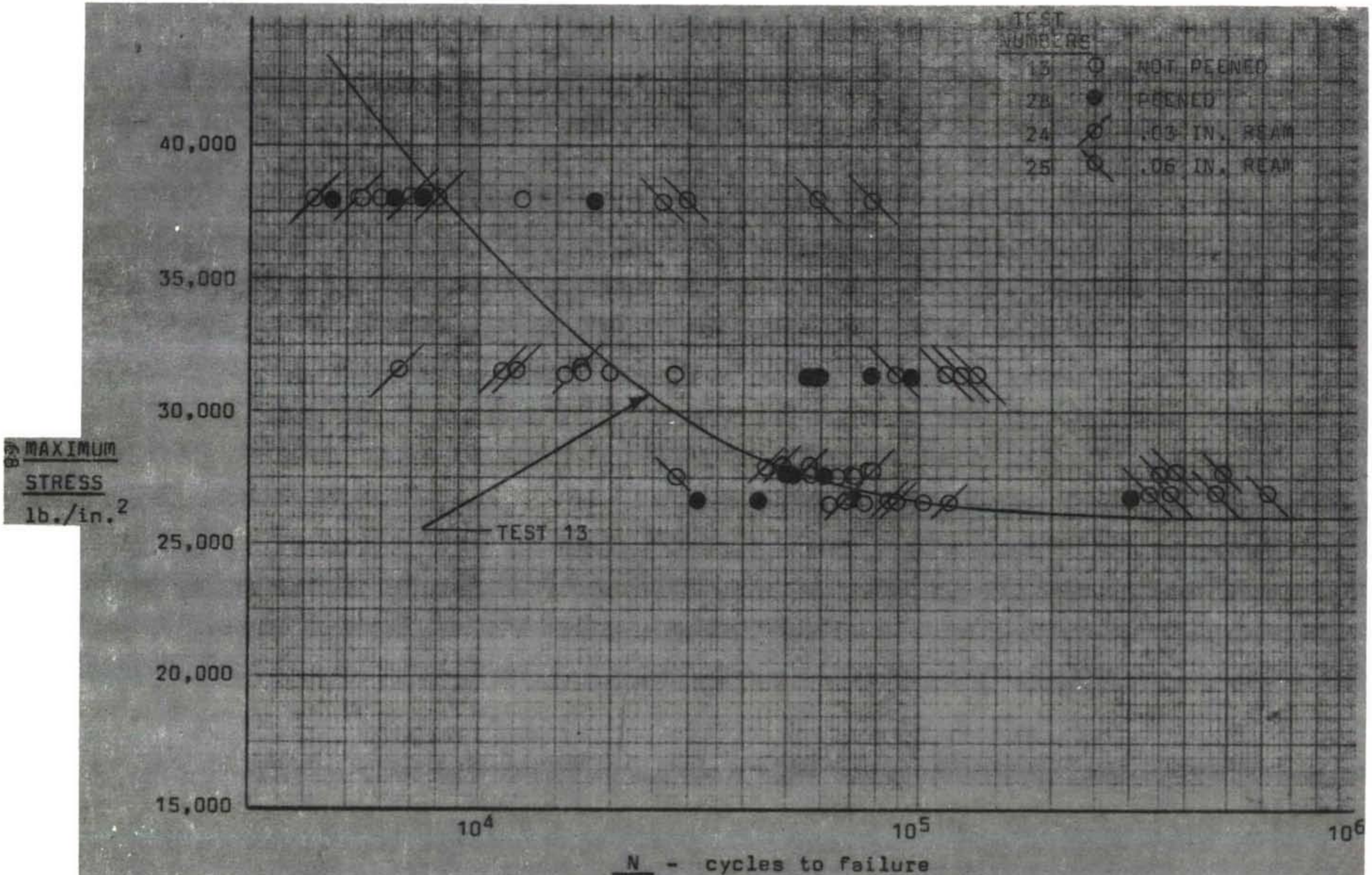

Figure A29. 4 Bolt Joint Specimen Test Results. Not Peened, Peened (without Torque Control), .03 in. and .06 in. Ream at 50% Life. 20,000 psi Mean Stress.

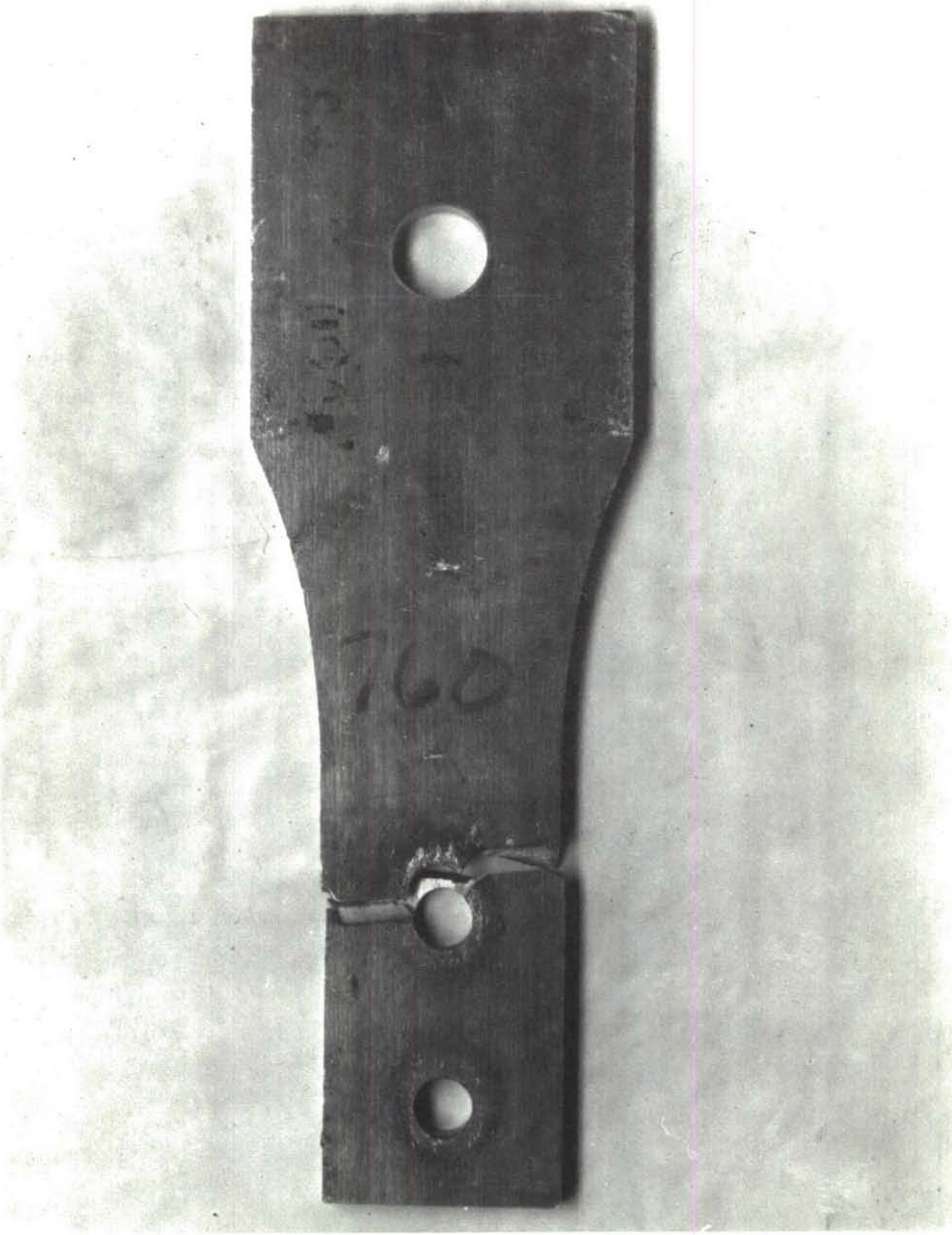

Figure 30. Failed 4 Bolt Joint Specimen.
Crack Initiated Outside of Hole Due to Bolt Preload.

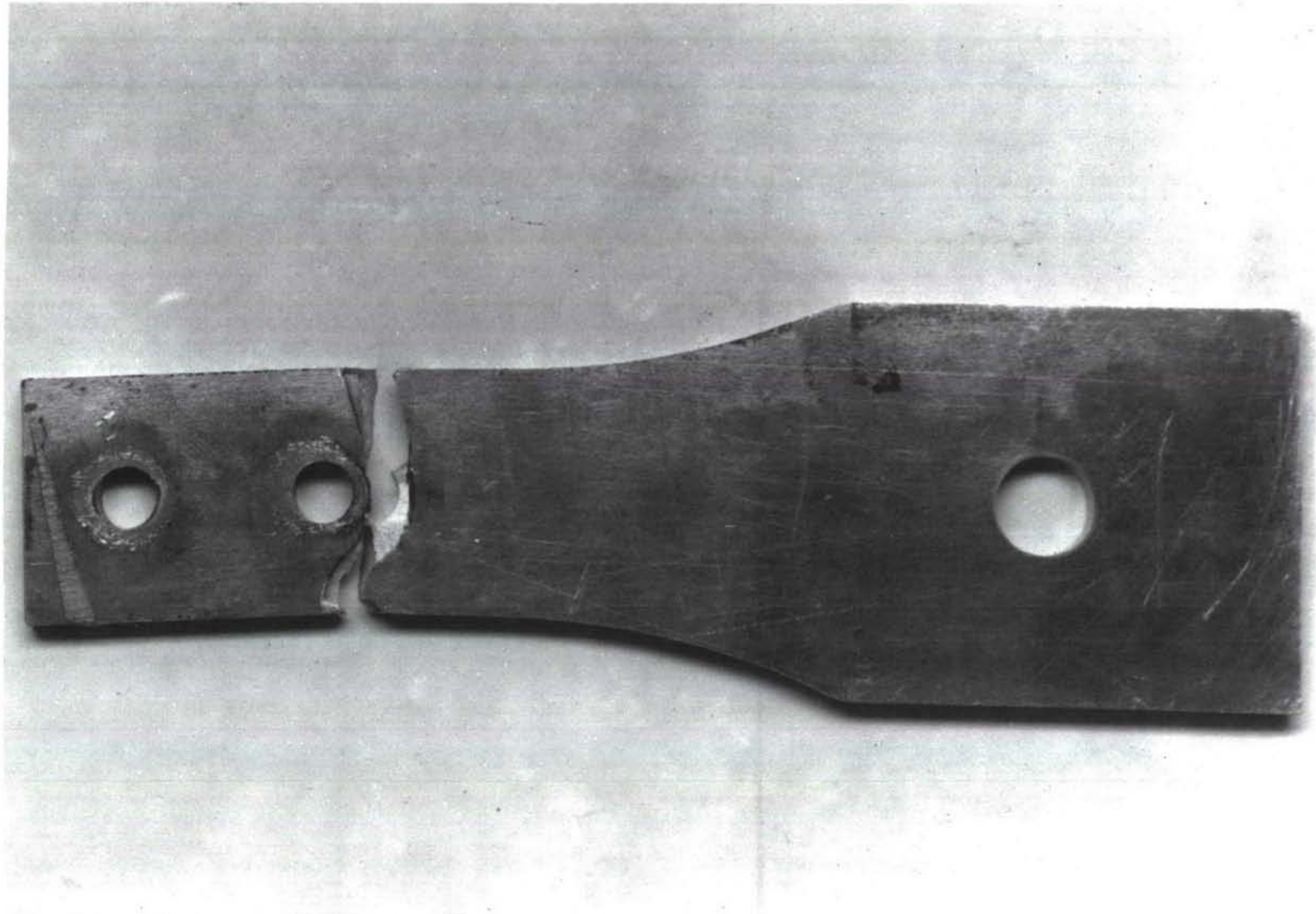

Figure 31. Failed 4 Bolt Joint Specimen.
Crack Initiated Outside of Hole Due to Bolt Preload.

APPENDIX II

SPECIMEN CYCLIC TEST DATA

LIST OF DATA SHEETS

LIST OF DATA SHEETS
(Continued)

LIST OF DATA SHEETS AND S-N CURVES BY TEST NUMBER

TEST NO.	DATA SHEET TABLE NO.	S-N CURVE FIGURE NO.	TEST NO.	DATA SHEET TABLE NO.	S-N CURVE FIGURE NO.
1	X	33	32	XLIX	69
2	IX	32	33	L	NONE
3	XI	34	34	XVIII	41
4	XII	35	35	XXXI	54
5	XIII	36	36	XXXV	58
6	XIV	37	37	XXXII	55
7	XVII	40	38	XIX	42
8	XVI	39	39	XL	NONE
9	XV	38	40	XXIX	52
10	XXXVII	60	41	LI	NONE
11	XXXVI	59	42	XLI	NONE
12	XLV	65	43	LII	NONE
13	XLIV	64	44	XLIII	NONE
14	XX	43	45	LIV	NONE
15	XXI	44	46	LV	71
16	XXII	45	47	LVII	73
17	XXIII	46	48	LVI	72
18	XXIV	47	49	L	NONE
19	XXV	48	50	XXVIII	51
20	XXVI	49			
21	XXVII	50			
22	XXXVIII	61			
23	XXXIX	62			
24	XLVI	66			
25	XLVII	67			
26	XXX	53			
27	XLII	63			
28	LIII	70			
29	XXXIV	57			
30	XXXIII	56			
31	XLVIII	68			

TABLE IX

TEST NUMBER 2 OPEN HOLE SPECIMENS

Open Hole Control
20,000 psi Mean Stress
Net Area, .270 in^2

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	459	>1,000,000	461	79,500	463	19,700	465	7,500
	460	>1,000,000	462	101,500	464	20,500	466	7,800
	70	84,500	77	49,300	79	23,400	81	7,800
	71	57,500	72	46,600	73	27,100	74	9,700
N (log mean)		69,200(70,71)		65,600		22,500		8,170
s (log)		------------		.164		.063		.051
Maximum Stress		25,600(459,460) 27,200(70, 71)		28,900		33,400		39,200
Mean Stress		19,100		19,200		19,600		19,600

TABLE X

TEST NUMBER 1 OPEN HOLE SPECIMENS

Open Hole Control.
10,000 psi Mean Stress.
Net Area, .272 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	452	267,200	453	100,400	455	49,700	457	14,200
	499	671,900	454	229,100	456	45,500	458	10,000
	500	588,100	469	180,700	471	48,900	473	7,000
	910	221,000	470	85,500	472	37,300	474	7,600
N (log mean)		391,000		137,500		45,100		9,340
s (log)		.242		.203		.058		.139
Maximum Stress		19,600		21,400		25,400		32,000
Mean Stress		9,770		9,700		9,740		9,700

TABLE XI

TEST NUMBER 3 OPEN HOLE SPECIMENS

Open Hole Control
.125 Inch Thickness
Net Area, .133 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	001	104,400	003	46,400	005	21,300	007	11,200
	002	108,500	004	59,900	006	22,400	008	8,300
	009C	66,400	011C	57,600	013C	18,400	015C	9,200
	010C	73,600	012C	41,100	014C	19,300	016C	8,300
N (log mean)		86,300		50,700		20,300		9,180
s (log)		.107		.077		.039		.061
Maximum Stress		28,300		30,600		34,300		40,000
Mean Stress		20,000		20,000		20,000		20,000

TABLE XII

TEST NUMBER 4 OPEN HOLE SPECIMENS

Open Hole Control.
Net Section, 90%
Net Area, .292 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	307	>1,000,000	309	70,800	311	17,500	313	15,200
	308	278,100	310	109,700	312	23,300	314	15,300
	315	115,800	317	61,000	319	25,100	321	10,100
	316	96,700	318	62,500	320	12,000	322	9,400
N (log mean)		146,000(308,315,216)		73,800		18,700		12,200
s (log)		.246		.118		.145		.113
Maximum Stress		25,400 (307) 26,900(308,315,316)		28,500		35,300		38,600
Mean Stress		19,100		19,300		19,300		19,300

TABLE XIII

TEST NUMBER 5 OPEN HOLE SPECIMENS

Open Hole Control.
Net Section, 75%.
Net Area, .245 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	923	709,500	293	65,200	297	24,800	295	17,600
	924	920,600	294	61,000	298	19,100	296	21,100
	299	1,141,000	301	59,300	305	22,300	303	9,500
	300	143,200	302	39,400	306	20,300	304	11,300
N (log mean)		572,000		55,200		21,500		14,100
s (log)		.410		.099		.049		.162
Maximum Stress		27,200		30,200		33,300		38,300
Mean Stress		19,300		19,200		19,100		19,200

TABLE XIV

TEST NUMBER 6 OPEN HOLE SPECIMENS

Open Hole Control.
Net Section, 65%.
Net Area, .214 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	259	>1,000,000	261	369,200	263	37,900	265	14,100
	260	121,800	262	124,400	264	28,700	266	13,900
	267	106,700	269	39,400	271	37,800	273	12,200
	268	111,400	270	88,500	272	34,100	274	12,500
N (log mean)		113,000(260,267,268)		113,000		34,400		13,200
s (log)		.029		.403		.057		.032
Maximum Stress		27,600		29,400		31,700		37,900
Mean Stress		18,900		18,900		19,000		18,900

TABLE XV

TEST NUMBER 9 **OPEN HOLE SPECIMENS**

Open Hole Control.
Edge Distance Ratio, 2.5
Net Area, .278 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	247	144,000	249	61,000	251	26,300	255	8,300
	248	126,900	250	79,600	252	27,500	256	9,900
	243	100,500	245	48,600	253	25,700	257	10,300
	244	101,400	246	45,500	254	25,600	258	8,400
N (log mean)		116,800		57,300		26,300		9,190
s (log)		.076		.110		.015		.048
Maximum Stress		27,300		28,900		32,400		38,100
Mean Stress		19,200		19,200		19,200		19,200

TABLE XVI

TEST NUMBER 8 OPEN HOLE SPECIMENS

Open Hole Control.
Edge Distance Ratio, 2.0
Net Area, .277 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	227	122,900	229	55,400	231	30,200	233	9,500
	228	73,000	230	65,300	232	32,200	234	7,100
	235	76,400	237	46,600	239	16,500	241	9,900
	236	201,400	238	47,000	240	21,500	242	11,400
N (log mean)		108,400		53,100		24,300		9,360
s (log)		.206		.069		.135		.087
Maximum Stress		27,300		28,900		32,600		38,200
Mean Stress		19,200		19,200		19,200		19,200

TABLE XVII

TEST NUMBER 7 OPEN HOLE SPECIMENS

Open Hole Control.
Edge Distance Ratio, 1.5
Net Area, .278 $in.^2$

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	211	67,000	213	50,100	215	18,300	217	6,200
	212	110,500	214	63,000	216	20,300	218	7,700
	219	77,800	221	38,900	223	13,400	225	6,200
	220	94,700	222	46,600	224	16,500	226	8,500
N (log mean)		85,900		49,000		17,000		7,080
s (log)		.095		.087		.077		.069
Maximum Stress		27,200		28,600		32,400		38,100
Mean Stress		19,100		19,000		19,100		19,100

TABLE XVIII

TEST NUMBER 34 OPEN HOLE SPECIMENS

Bare Material Control.
Net Area, .263 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	339	110,600	343	58,600	347	25,500	351	11,800
	340	198,900	344	92,900	348	28,500	352	10,700
	341	464,700	345	69,300	349	33,100	353	11,000
	342	>1,000,000	346	422,900	350	31,000	354	13,200
N (log mean)		217,000		112,100		29,400		11,650
s (log)		.314		.392		.0487		.0387
Maximum Stress		28,700		30,500		34,100		40,100
Mean Stress		20,200		20,200		20,300		20,300

TABLE XIX

TEST NUMBER 38 OPEN HOLE SPECIMENS

Control, Open Hole, Surface
Covered with Grease.
Hole Diameter, .312 in.
Net Area, .248 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	387	89,500	391	45,200	395	26,700	399	10,900
	388	130,700	392	56,200	396	32,900	400	10,700
	389	>1,000,000	393	61,800	397	30,000	401	12,300
	390	743,600	394	45,600	398	22,500	402	11,600
N (log mean)		206,000		51,800		27,800		11,350
s (log)		.490		.0667		.071		.0274
Maximum Stress		26,700		29,900		33,000		37,800
Mean Stress		19,000		19,000		18,900		18,900

TABLE XX

TEST NUMBER 14 OPEN HOLE SPECIMENS

Open Hole Reaming.
Hole Diameter Increase, .03 in.
Ream at 25-66% Life.
Net Area Before Reaming, .277 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	116	435,600	119	86,400	123	29,300	127	10,200
	117	229,600	120	142,300	124	31,300	128	9,300
			121	85,900	125	30,300	129	9,100
			122	99,500	126	27,600	130	13,500
N (log mean)		316,000		101,000		29,600		10,380
s (log)		.197		.104		.0231		.0786
Maximum Stress		27,900		28,900		32,500		38,100
Mean Stress		19,200		19,100		19,100		19,000
N (Ream Cycles)		19,700		20,000		6,000		2,000
N (Net Ream Cycles)		296,300		81,000		23,600		8,380
Actual % Life		17.9		29.0		21.4		19.9

TABLE XXI

TEST NUMBER 15 OPEN HOLE SPECIMENS

Open Hole Reaming.
Hole Diameter Increase, .03 in.
Ream at 25-65% Life.
Net Area Before Reaming, .280 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	118	194,000	135	107,900	139	48,500	143	12,700
	132	214,400	136	97,200	140	45,200	144	13,600
			137	107,300	141	39,700	145	11,100
			138	120,200	142	36,500	146	14,400
N (log mean)		204,000		107,800		42,200		12,900
s (log)		.0297		.0376		.0545		.0488
Maximum Stress		27,500		28,600		32,100		38,000
Mean Stress		18,900		18,900		18,900		19,000
N (Ream Cycles)		39,300		33,000		10,000		3,300
N (Net Ream Cycles)		164,700		74,800		32,200		9,600
Actual % Life		29.1		41.3		33.3		32.1

TABLE XXII

TEST NUMBER 16 OPEN HOLE SPECIMENS

Open Hole Reaming.
Hole Diameter Increase, .03 in.
Ream at 25%-66% Life.
Net Area Before Reaming, .280 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	133	60,500	151	92,900	155	44,700	159	14,900
	134	199,100	152	153,200	156	39,500	160	13,500
			153	112,200	157	36,900	161	13,300
			154	116,000	158	38,100	162	13,300
N (log mean)		110,000		116,500		39,600		13,700
s (log)		.365		.0895		.0363		.0237
Maximum Stress		27,500		28,700		32,100		37,900
Mean Stress		18,900		19,000		18,900		18,900
N (Ream Cycles)		59,000		43,000		12,900		4,300
N (Net Ream Cycles)		51,000		93,500		26,700		9,400
Actual % Life		43.7		55.2		43.0		41.0

TABLE XXIII

TEST NUMBER 17 OPEN HOLE SPECIMENS

Open Hole Reaming.
Hole Diameter Increase, .03 in.
Ream at 100% Life (Initial Crack).
Net Area Before Reaming, .278 in.^{2}

	Specimen Number	N	(Initial Crack) N	(Net Ream Cycles) N	Specimen Number	N	(Initial Crack) N	(Net Ream Cycles) N
Specimen	83C	83,300	82,200	1,100	87C	53,500	50,800	2,700
Number	84C	73,100	70,800	2,300	88C	47,500	43,400	4,100
and Cycles	85C	81,600	75,100	6,500	89C	48,300	43,400	4,900
to Failure	86C	98,000	86,800	11,200	90C	52,500	45,900	6,600
N (log mean)		83,500				50,300		
s (log)		.0524				.0258		
Maximum Stress		27,700				28,800		
Mean Stress		19,100				19,100		
Specimen	93C	23,800	23,100	700	103C	8,700	8,400	1,100
Number	94C	24,700	24,000	700	104C	8,800	8,400	2,300
and Cycles	95C	19,500	18,400	1,100	105C	8,300	8,000	6,500
to Failure	101C	29,400	29,200	200	106C	8,100	7,700	11,200
N (log mean)		24,000				8,450		
s (log)		.0731				.0172		
Maximum Stress		32,000				38,100		
Mean Stress		19,100				19,100		

TABLE XXIV

TEST NUMBER 18 OPEN HOLE SPECIMENS

Open Hole Reaming.
Hole Diameter Increase, .06 in.
Ream at 25-66% Life.
Net Area Before Reaming, .279 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	163	1,039,900	167	185,400	171	33,700	175	12,400
	164	127,100	168	150,600	172	40,700	176	12,000
	165	189,400	169	306,400	173	43,700	177	12,100
	166	363,900	170	121,100	174	41,800	178	10,800
N (log mean)		309,000		179,000		39,800		11,700
s (log)		.398		.172		.0496		.0265
Maximum Stress		27,700		28,600		32,200		37,900
Mean Stress		19,100		18,900		19,000		18,900
N (Ream Cycles)		19,700		20,000		6,000		2,000
N (Net Ream Cycles		289,300		159,000		33,800		9,700
Actual % Life		15.8		25.0		20.7		19.0

TABLE XXV

TEST NUMBER 19 OPEN HOLE SPECIMENS

Open Hole Reaming.
Hole Diameter Increase, .06 in.
Ream at 25-66% Life.
Net Area Before Reaming, .279 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	179	80,300	183	226,600	187	36,100	191	12,100
	180	181,100	184	88,500	188	41,600	192	14,900
	181	78,300	185	184,200	189	35,300	193	14,500
	182	112,600	186	113,300	190	38,800	194	16,900
N (log mean)		106,100		143,000		37,800		14,500
s (log)		.170		.187		.0316		.0597
Maximum Stress		27,700		28,600		32,300		38,100
Mean Stress		19,100		19,100		19,100		19,100
N (Ream Cycles)		39,300		33,000		10,000		3,300
N (Net Ream Cycles)		66,800		110,000		27,800		11,200
Actual % Life		32.8		41.3		34.4		33.0

TABLE XXVI

TEST NUMBER 20 OPEN HOLE SPECIMENS

Open Hole Reaming.
Hole Diameter Increase, .06 in.
Ream at 25-66% Life.
Net Area Before Reaming, .279 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	195	30,600	199	112,900	203	49,100	207	14,800
	196	173,800	200	102,400	204	43,800	208	13,800
	197	150,100	201	100,600	205	55,000	209	14,100
	198	106,200	202	130,700	206	44,400	210	15,800
N (log mean)		96,000		111,000		48,000		14,600
s (log)		.342		.052		.046		.026
Maximum Stress		27,600		28,700		32,400		38,000
Mean Stress		19,000		19,000		19,100		19,000
N (Ream Cycles)		59,000		43,000		12,900		4,300
N (Net Ream Cycles)		37,000		68,000		35,100		10,300
Actual % Life		45.4		53.6		46.0		41.8

TABLE XXVII

TEST NUMBER 21 OPEN HOLE SPECIMENS

Open Hole Reaming.
Hole Diameter Increase, .06 in.
Ream at 100% Life (Initial Crack)
Net Area Before Reaming, .277 in.^{2}

	Specimen Number	N	(Initial Crack) N	(Net Ream Cycles) N	Specimen Number	N	(Initial Crack) N	(Net Ream Cycles) N
Specimen	107	77,500	72,400	5,100	111	93,000	88,900	4,100
Numbers	108	86,600	79,400	7,200	112	54,300	49,200	5,100
and Cycles	109	83,400	80,100	3,300	113	64,500	59,600	4,900
to Failure	110	81,700	74,300	7,400	114	94,200		
N (log mean)		82,100				74,400		
s (log)		.020				.119		
Maximum Stress		27,800				28,900		
Mean Stress		19,100				19,100		
Specimen	52	23,700	23,100	600	78	8,200	7,900	300
Numbers	69	22,800	22,500	300	80	7,900	7,700	200
and Cycles	75	22,800	21,200	1,600	82	8,300	8,100	200
to Failure	76	22,500	21,900	600	800	8,300	8,100	200
N (log mean)		22,900				8,160		
s (log)		.010				.010		
Maximum Stress		32,600				38,300		
Mean Stress		19,200				19,100		

TABLE XXVIII

TEST NUMBER 50   OPEN HOLE SPECIMENS

Short Edge Distance, Double Ream.
Edge Distance Ratio Before Reaming, 2.5.
Hole Diameter Increase: .06 in. at 50% Life,
.03 in. at 100% Life.
Net Area Before Reaming, .280 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	435	130,900	439	22,600	443	29,100	447	3,800
	436	44,000	440	------	444	20,500	448	3,900
	437	128,700	441	31,500	445	28,500	449	3,500
	438	133,700	442	82,700	446	21,800	450	3,600
N (log mean)		99,800		38,900		24,500		3,700
s (log)		.237		.293		.078		.030
Maximum Stress		27,000		28,800		32,100		38,000
Mean Stress		18,950		19,100		18,950		18,950
N_1		44,500		24,500		8,500		3,500
N_2		44,500		14,400		8,500		200
N_3		10,800		--		7,500		--

TABLE XXIX

TEST NUMBER 40 OPEN HOLE SPECIMENS

Short Edge Distance, Double Ream.
Edge Distance Ratio Before Reaming, 1.5.
Hole Diameter Increase: .06 in. at 50% Life,
.03 in. at 100% Life.
Net Area Before Reaming, .278 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	419	81,700	423	62,500	427	27,000	431	9,300
	420	83,800	424	87,000	428	31,500	432	8,100
	421	99,700	425	34,300	429	22,000	433	7,600
	422	79,200	426	61,800	430	22,700	434	8,200
N (log mean)		85,700		58,300		25,500		8,250
s (log)		.0446		.168		.0727		.0364
Maximum Stress		27,200		28,800		32,400		38,100
Mean Stress		19,100		19,100		19,100		19,100
N_1		44,500		24,500		8,500		3,500
N_2		41,200		24,500		8,500		3,500
N_3		--		9,300		8,500		1,250

TABLE XXX

TEST NUMBER 26 OPEN HOLE SPECIMENS

Open Hole Peening (.007A).
Net Area, .278 $in.^2$

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	275	83,400	279	67,800	283	44,200	287	10,900
	276	86,800	280	69,300	284	39,300	288	11,100
	277	73,000	281	67,300	285	33,700	289	10,000
	278	82,300	282	59,500	286	30,400	290	9,700
N (log mean)		81,000		65,800		36,500		10,400
s (log)		.0312		.0302		.0718		.0285
Maximum Stress		27,700		28,800		32,400		38,200
Mean Stress		19,100		19,100		19,100		19,100

TABLE XXXI

TEST NUMBER 35 OPEN HOLE SPECIMENS

Open Hole Peening, Bare Material (.007A).
Net Area, .265 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	355	61,500	359	52,100	363	19,700	367	8,200
	356	48,300	360	55,500	364	19,600	368	8,600
	357	44,600	361	60,600	365	23,600	369	8,700
	358	65,900	362	52,700	366	22,600	370	8,000
N (log mean)		54,400		55,000		21,300		8,360
s (log)		.0813		.031		.0415		.0172
Maximum Stress		29,100		30,100		34,000		39,800
Mean Stress		20,000		20,000		20,000		20,000

TABLE XXXII

TEST NUMBER 37 OPEN HOLE SPECIMENS

Open Hole Peening (.007A),
.250 in. Hole.
Net Area, .263 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	371	95,800	375	73,200	379	29,400	383	9,900
	372	96,300	376	72,500	380	27,500	384	9,600
	373	91,700	377	82,000	381	25,600	385	10,100
	374	95,900	378	61,500	382	31,200	386	9,900
N (log mean)		95,000		72,000		28,300		9,860
s (log)		.0097		.0515		.0369		.0092
Maximum Stress		27,500		28,400		33,300		38,000
Mean Stress		19,000		19,000		19,000		19,000

TABLE XXXIII

TEST NUMBER 30 OPEN HOLE SPECIMENS

Peening on Clad Material.
High Intensity Peening, Outside of Hole.
(Steel Shot, .015A).
Net Area, .261 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	639	131,300	643	83,800	647	29,500	651	12,000
	640	126,300	644	84,700	648	31,500	652	14,800
	641	126,300	645	88,100	649	31,100	653	17,000
	642	110,200	646	94,000	650	29,700	654	14,600
N (log mean)		123,200		87,500		30,400		14,450
s (log)		.034		.020		.0141		.062
Maximum Stress		27,900		28,800		33,400		38,000
Mean Stress		19,200		19,200		19,100		19,000

TABLE XXXIV

TEST NUMBER 29 OPEN HOLE SPECIMENS

Peening on Bare Material, .250 in. Hole.
High Intensity Peening, Outside of Hole.
(Steel Shot, .015A).
Net Area, .250 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	623	81,400	627	72,500	631	21,600	635	7,500
	624	99,100	628	71,300	632	23,000	636	9,800
	625	134,800	629	66,800	633	22,100	637	9,600
	626	99,900	630	83,100	634	23,400	638	8,400
N (log mean)		102,000		73,200		22,400		8,760
s (log)		.091		.0403		.0161		.054
Maximum Stress		28,900		30,100		35,200		40,000
Mean Stress		19,900		20,100		20,100		20,000

TABLE XXXV

TEST NUMBER 36 OPEN HOLE SPECIMENS

Open Hole, High Intensity Peening, Outside of Hole (.015A).
Specimen Thickness, .125 in.
Net Area, .133 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	17	93,700	21	74,200	25	21,200	29	11,000
	18	90,800	22	77,500	26	22,700	30	11,500
	19	81,200	23	60,900	27	25,300	31	8,000
	20	89,500	24	60,100	28	26,800	32	10,600
N (log mean)		88,700		67,900		23,900		10,180
s (log)		.0244		.0572		.0459		.0709
Maximum Stress		28,200		30,500		34,200		39,000
Mean Stress		19,900		19,900		19,900		19,500

TABLE XXXVI

TEST NUMBER 11 LOADED HOLE SPECIMENS

Loaded Hole Control.
20,000 psi Mean Stress.
Net Area, .260 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	535	118,500	539	98,100	543	32,600	547	8,500
	536	266,300	540	124,600	544	21,100	548	9,200
	537	170,700	541	73,400	545	20,100	549	12,100
	538	126,300	542	97,500	546	43,100	550	8,900
N (log mean)		162,000		96,800		27,800		9,600
s (log)		.161		.094		.158		.069
Maximum Stress		27,800		28,800		33,600		38,500
Mean Stress		19,200		19,200		19,200		19,200

TABLE XXXVII

TEST NUMBER 10 LOADED HOLE SPECIMENS

Loaded Hole Control.
10,000 psi Mean Stress.
Net Area, .262 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	523	123,700	527	54,700	531	29,000	510	4,500
	524	74,500	528	93,000	532	25,500	511	6,200
	525	420,200	529	59,500	533	28,700	521	4,200
	526	223,900	530	78,600	534	16,600	522	4,600
N (log mean)		171,500		69,600		24,300		4,810
s (log)		.325		.1065		.114		.0746
Maximum Stress		21,100		25,700		30,400		32,500
Mean Stress		9,590		9,500		9,500		9,590

TABLE XXXVIII

TEST NUMBER 22 LOADED HOLE SPECIMENS

Loaded Hole Reaming.
Hole Diameter Increase, .03 in.
Ream at 50% Life.
Net Area Before Reaming, .259 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	551	143,900	555	140,000	559	27,200	563	12,200
	552	131,800	556	97,000	560	37,400	564	15,100
	553	137,000	557	85,900	561	33,300	565	16,200
	554	161,200	558	193,800	562	39,300	566	24,000
N (log mean)		143,000		122,500		33,900		16,320
s (log)		.038		.160		.071		.123
Maximum Stress		28,100		29,000		33,800		38,500
Mean Stress		19,300		19,300		19,300		19,200
N (Ream Cycles)		81,000		48,400		13,900		4,800
N (Net Ream Cycles)		62,000		74,100		20,000		11,520

TABLE XXXIX

TEST NUMBER 23 LOADED HOLE SPECIMENS

Loaded Hole Reaming.
Hole Diameter Increase, .06 in.
Ream at 50% Life.
Net Area Before Reaming, .262 $in.^2$

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	567	403,700	571	259,900	575	95,600	579	41,200
	568	374,700	572	259,200	576	213,600	580	43,900
	569	534,200	573	533,400	577	34,600	605	19,800
	570	654,000	574	402,000	578	31,900	606	33,100
N (log mean)		480,000		347,000		68,900		33,000
s (log)		.111		.154		.392		.1575
Maximum Stress		27,600		28,700		33,400		38,300
Mean Stress		19,000		19,000		19,000		19,000
N (Ream Cycles)		81,000		48,400		13,900		4,800
N (Net Ream Cycles)		399,000		298,600		55,000		28,200

TABLE XL

TEST NUMBER 39 LOADED HOLE SPECIMENS

Grease Between Specimen and Doubler Plates.
Net Area, .262 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	607	34,100	611	34,100	615	19,700	619	18,100
	608	19,100	612	46,000	616	23,200	620	19,000
	609	30,700	613	43,800	617	22,600	621	19,000
	610	25,800	614	39,500	618	17,100	622	18,900
N (log mean)		26,800		40,500		20,500		18,700
s (log)		.1098		.0573		.0609		.0106
Maximum Stress		33,400		33,400		33,400		33,400
Mean Stress		19,100		19,100		19,100		19,100
Ream, 50% Life, in.		0		0		.03		.06
Bolt Torque, in.-lb.		0		25		0		0

TABLE XLI

TEST NUMBER 42 LOADED HOLE SPECIMENS

Loaded Hole, Bolt Preload Variation.
Net Area Before Reaming, .261 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	703	24,900	707	29,100	711	37,400	715	138,000
	704	19,200	708	18,800	712	27,400	716	89,500
	705	18,900	709	22,100	713	36,300	717	71,600
	706	19,300	710	35,600	714	39,000	718	45,000
N (log mean)		20,400		25,600		37,400		79,500
s (log)		.0573		.1235		.0699		.203
Maximum Stress		33,400		33,400		33,500		33,600
Mean Stress		19,100		19,100		19,200		19,300
Hole Dia., in.		.248		.248		.312		.312
Bolt Torque, in.-lb.		0		20		45		70

TABLE XLII

TEST NUMBER 27 LOADED HOLE SPECIMENS

Loaded Hole Peening (.007A)
Net Area, .263 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	599	416,800	603	>1,000,000	695	39,800	699	13,700
	600	972,500	604	647,500	696	150,200	700	39,800
	601	712,100	693	523,900	697	100,300	701	30,900
	602	>1,000,000	694	263,500	698	19,300	702	51,800
N (log mean)		660,000(599,600,601)		447,000(604,693,694)		58,400		30,600
s (log)		-------		-------		.412		.250
Maximum Stress		27,600		28,500		33,200		38,000
Mean Stress		19,000		19,000		19,000		19,000

TABLE XLIII

TEST NUMBER 44 LOADED HOLE SPECIMENS

Loaded Hole, Peened, Bolt Preload Variation.
Net Area Before Reaming, .261 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	719	22,000	723	27,900	727	36,100	731	41,600
	720	22,700	724	27,400	728	51,300	732	38,500
	721	24,000	725	19,800	729	27,100	733	37,300
	722	19,300	726	22,800	730	48,500	734	38,300
N (log mean)		21,900		24,200		39,500		38,900
s (log)		.0398		.0707		.128		.0207
Maximum Stress		31,900		31,800		31,500		31,400
Mean Stress		19,300		19,200		19,100		19,000
Hole Dia., in.		.248		.248		.312		.312
Bolt Torque, in.-lb.		0		20		45		70

TABLE XLIV

TEST NUMBER 13   4 BOLT JOINT

4 Bolt Joint Control.
20,000 psi Mean Stress.
Net Area, .263 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	661	74,700	665	64,800	669	17,100	673	12,400
	662	67,300	666	56,800	670	27,900	674	6,000
	663	102,000	667	71,100	671	15,800	675	7,500
	664	63,200	668	70,000	672	19,700	676	5,500
N (log mean)		75,500		65,500		19,650		7,450
s (log)		.092		.045		.110		.158
Maximum Stress		26,600		27,600		31,400		38,100
Mean Stress		19,000		19,000		19,000		19,100

TABLE XLV

TEST NUMBER 12 4 BOLT JOINT

4 Bolt Joint Control.
10,000 psi Mean Stress.
Net Area, .262 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	503	208,000	513	93,900	517	23,700	657	8,000
	504	187,100	514	57,500	518	52,600	658	8,000
	505	289,200	515	65,500	655	21,100	659	7,300
	506	255,300	516	142,400	656	23,000	660	7,000
N (log mean)		231,000		84,300		28,000		7,570
s (log)		.085		.177		.185		.029
Maximum Stress		19,000		20,100		23,800		30,600
Mean Stress		9,520		9,570		9,540		9,540

TABLE XLVI

TEST NUMBER 24 4 BOLT JOINT SPECIMENS

4 Bolt Joint Reaming.
Hole Diameter Increase, .03 in.
Ream at 50% Life.
Net Area Before Reaming, .262 $in.^2$

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	677	89,500	681	56,400	685	20,000	689	4,200
	678	110,800	682	44,600	686	16,300	690	8,100
	679	85,200	683	48,600	687	21,000	691	5,400
	680	67,400	684	79,700	688	26,800	692	7,000
N (log mean)		86,600		55,900		20,600		5,980
s (log)		.089		.111		.089		.126
Maximum Stress		26,600		27,700		31,600		38,100
Mean Stress		19,000		19,100		19,100		19,100
N (Ream Cycles)		34,200		31,900		9,800		4,000
N (Net Ream Cycles)		52,400		24,000		10,800		1,980

TABLE XLVII

TEST NUMBER 25 4 BOLT JOINT SPECIMENS

4 Bolt Joint Reaming.
Hole Diameter Increase, .06 in.
Ream at 50% Life.
Net Area Before Reaming, .263 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	749	615,300	753	487,600	757	88,300	761	79,400
	750	466,200	754	380,700	758	134,000	762	59,100
	751	331,000	755	356,700	759	115,800	763	29,600
	752	374,700	756	28,200	760	125,700	764	26,100
N (log mean)		433,000		208,000		114,000		43,500
s (log)		.117		.581		.0796		.234
Maximum Stress		26,700		27,600		31,400		38,000
Mean Stress		19,100		19,000		19,000		19,000
N (Ream Cycles)		34,200		31,900		9,800		4,000
N (Net Ream Cycles)		398,800		176,100		104,200		39,500

TABLE XLVIII

TEST NUMBER 31 4 BOLT JOINT SPECIMENS

4 Bolt Joint Reaming, Low Bolt Preload.
Hole Diameter Increase, .03 in.
Bolt Torque, 12 inch-pounds.
Ream at 50% Life.
Net Area, .262 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	765	65,000	769	30,500	773	21,000	777	6,200
	766	76,600	770	45,400	774	28,900	778	6,300
	767	69,400	771	27,200	775	28,000	779	10,000
	768	73,000	772	53,900	776	24,100	780	10,300
N (log mean)		71,000		37,600		25,200		7,950
s (log)		.0282		.140		.0638		.122
Maximum Stress		26,700		27,600		31,400		38,200
Mean Stress		19,100		19,000		19,000		19,100
N (Ream Cycles)		34,200		31,900		9,800		4,000
N (Net Ream Cycles)		36,800		5,700		15,400		3,950

TABLE XLIX

TEST NUMBER 32 4 BOLT JOINT SPECIMENS

4 Bolt Joint Reaming, Low Bolt Preload.
Hole Diameter Increase, .06 in.
Bolt Torque, 12 inch-pounds.
Ream at 50% Life.
Net Area, .263 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	781	103,900	785	72,700	789	31,400	793	6,900
	782	239,100	786	69,300	790	22,900	794	11,500
	783	118,900	787	53,500	791	26,300	795	11,800
	784	86,500	788	97,800	792	33,100	796	8,900
N (log mean)		126,100		71,600		28,100		9,520
s (log)		.195		.08		.0729		.1092
Maximum Stress		26,500		27,600		31,400		38,000
Mean Stress		18,900		19,000		19,000		19,000
N (Ream Cycles)		34,200		31,900		9,800		4,000
N (Net Ream Cycles)		91,900		39,700		18,300		5,520

TABLE L

TEST NUMBER 33 4 BOLT JOINT SPECIMENS

Miscellaneous Reaming and Bolt Preloads.
Net Area, .263 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	797	------	801	71,200	805	25,800	809	37,500
	798	12,900	802	81,600	806	24,900	810	41,900
	799	13,200	803	73,600	807	21,600	811	38,600
	800	14,200	804	46,500	808	27,700	812	37,900
N (log mean)		13,400		66,600		24,900		34,700
s (log)		.022		.107		.0455		.108
Maximum Stress		31,300		31,300		31,300		31,300
Mean Stress		19,000		19,000		19,000		19,000
Ream Increase, in.		.06		.06		.03		.06
Ream, % Life		0		50		50		50
Bolt Torque, in.-lb.		0		45		35		70

TABLE LI

TEST NUMBER 41 4 BOLT JOINT SPECIMENS

4 Bolt Joint, Grease Between Specimen and Splice Plates.
Zero Bolt Torque.
Net Area Before Reaming, .263 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	813	19,700	817	12,200	821	------	825	16,300
	814	18,300	818	13,900	822	14,900	826	26,300
	815	13,300	819	12,800	823	9,700	827	27,000
	816	14,600	820	12,800	824	12,300	828	25,400
N (log mean)		16,300		12,900		12,100		23,300
s (log)		.0801		.0236		.093		.1038
Maximum Stress		31,400		31,400		31,400		31,400
Mean Stress		19,000		19,000		19,000		19,000
Initial Hole Dia., in.		.248		.253		.248		.248
Ream at 50% Life, Hole Dia. Increase, in.		0		0		.03		.06

TABLE LII

TEST NUMBER 43   4 BOLT JOINT SPECIMENS

4 Bolt Joint, Bolt Preload Variation.
Net Area, .262 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	829	17,800	833	22,900	837	29,500	841	93,500
	830	14,600	834	18,000	838	51,500	842	160,800
	831	12,100	835	15,100	839	64,000	843	347,900
	832	17,400	836	24,400	840	36,500	844	125,300
N (log mean)		15,300		19,700		43,400		160,000
s (log)		.0776		.0963		.150		.244
Maximum Stress		33,100		33,100		33,100		33,100
Mean Stress		20,100		20,100		20,100		20,100
Hole Dia., in.		.248		.248		.312		.312
Bolt Torque, in.-lb.		0		20		45		70

TABLE LIII

TEST NUMBER 28 4 BOLT JOINT SPECIMENS

4 Bolt Joint Peening (.007A).
Net Area, .263 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	583	301,700	587	61,600	591	95,800	595	18,200
	584	70,500	588	51,100	592	59,600	596	7,500
	585	43,100	589	52,100	593	56,900	597	4,700
	586	31,800	590	50,800	594	77,900	598	6,400
N (log mean)		73,500		53,700		71,000		8,000
s (log)		.433		.040		.1053		.252
Maximum Stress		26,700		27,600		31,300		38,000
Mean Stress		19,100		19,000		19,000		19,000

TABLE LIV

TEST NUMBER 45 4 BOLT JOINT SPECIMENS

4 Bolt Joint, Peened,
Bolt Preload Variation.
Net Area Before Reaming, .263 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	845	6,200	849	8,200	853	11,600	857	24,300
	846	7,400	850	9,700	854	14,800	858	15,800
	847	7,200	851	9,400	855	10,900	859	19,000
	848	6,100	852	8,800	856	11,300	860	22,000
N (log mean)		6,700		9,000		12,050		20,000
s (log)		.0433		.0323		.0603		.0817
Maximum Stress		33,400		33,400		33,400		33,400
Mean Stress		19,000		19,000		19,000		19,000
Hole Dia., in.		.248		.248		.312		.312
Bolt Torque, in.-lb.		0		20		45		70

TABLE LV

TEST NUMBER 46 — 8 BOLT JOINT SPECIMENS

8 Bolt Joint Control.
20,000 psi Mean Stress.
Net Area, .219 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	863	142,600	962	251,100	871	54,600	875	9,800
	864	136,400	963	266,000	872	84,200	960	29,300
	865	174,600	869	120,900	873	56,000	961	17,400
	866	161,000	870	167,500	874	56,100	878	10,200
N (log mean)		152,800		191,900		61,700		15,000
s (log)		.049		.161		.090		.224
Maximum Stress		25,800		26,600		30,100		36,600
Mean Stress		18,300		18,300		18,300		18,300

TABLE LVI

TEST NUMBER 48 8 BOLT JOINT SPECIMENS

10,000 psi Mean Stress Control.
Net Area, .220 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	895	264,800	899	132,100	952	92,900	956	37,800
	896	342,600	900	214,900	953	109,800	957	33,200
	897	158,000	950	204,500	954	72,700	958	30,300
	898	219,300	951	121,100	955	133,200	959	52,900
N (log mean)		237,000		163,000		99,500		37,700
s (log)		.142		.128		.111		.106
Maximum Stress		19,100		22,500		24,900		31,200
Mean Stress		9,530		9,610		9,500		9,500

TABLE LVII

TEST NUMBER 47 8 BOLT JOINT SPECIMENS

Hole Diameter Increase, .06 in.
Ream and Peen at 50% Life.
Net Area Before Reaming, .220 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	879	>1,000,000	883	315,900	887	415,900	891	28,300
	880	386,800	884	327,100	888	591,700	892	81,000
	881	307,600	885	222,700	889	412,400	893	63,400
	882	575,600	886	293,800	890	500,000	894	53,100
N (log mean)		409,000		286,500		474,000		52,700
s (log)		.138		.075		.0745		.195
Maximum Stress		26,700		27,700		31,400		37,900
Mean Stress		19,000		19,100		19,100		19,000
N (Ream Cycles)		125,000		75,000		28,000		7,500
N (Net Ream Cycles)		284,000		211,500		446,000		45,200

TABLE LVIII

TEST NUMBER 49 8 BOLT JOINT SPECIMENS

Hole Diameter Increase, .06 in. Ream and Peen at 50% Life.
Net Area Before Reaming, .220 in.^{2}

	Specimen Number	N	Specimen Number	N	Specimen Number	N	Specimen Number	N
Specimen Numbers and Cycles to Failure	501	391,400	505	259,100	509	236,400	513	301,100
	502	429,200	506	252,400	510	252,300	514	274,900
	503	351,000	507	247,900	511	253,900	515	245,200
	504	375,500	508	254,900	512	240,700	516	260,800
N (log mean)		386,000		254,000		246,000		270,000
s (log)		.037		.008		.015		.038
N_1		15,400		24,900		3,800		59,300
Maximum Stress		31,400		25,000		38,200		19,100
Mean Stress		19,100		9,550		19,100		9,550
N_2		38,200		59,300		59,300		3,800
Maximum Stress		26,800		19,100		19,100		38,200
Mean Stress		19,100		9,550		9,550		19,100
Ream and Peen								
N_3		24,900		15,400		3,800		59,300
Maximum Stress		25,000		31,400		38,200		19,100
Mean Stress		9,550		19,100		19,100		9,550
N_4		59,300		38,200		59,300		3,800
Maximum Stress		19,100		26,800		19,100		38,200
Mean Stress		9,550		19,100		9,550		19,100
N_5		40,700		48,000		3,800		59,300
Maximum Stress		22,300		27,700		38,200		19,100
Mean Stress		9,550		19,100		19,100		9,550
N_6		207,500		68,200		59,300/56,700		84,500
Maximum Stress		31,400		38,200		19,100/38,200		38,200
Mean Stress		9,550		19,100		9,550/19,100		19,100

APPENDIX III

INDIVIDUAL TEST S-N CURVES

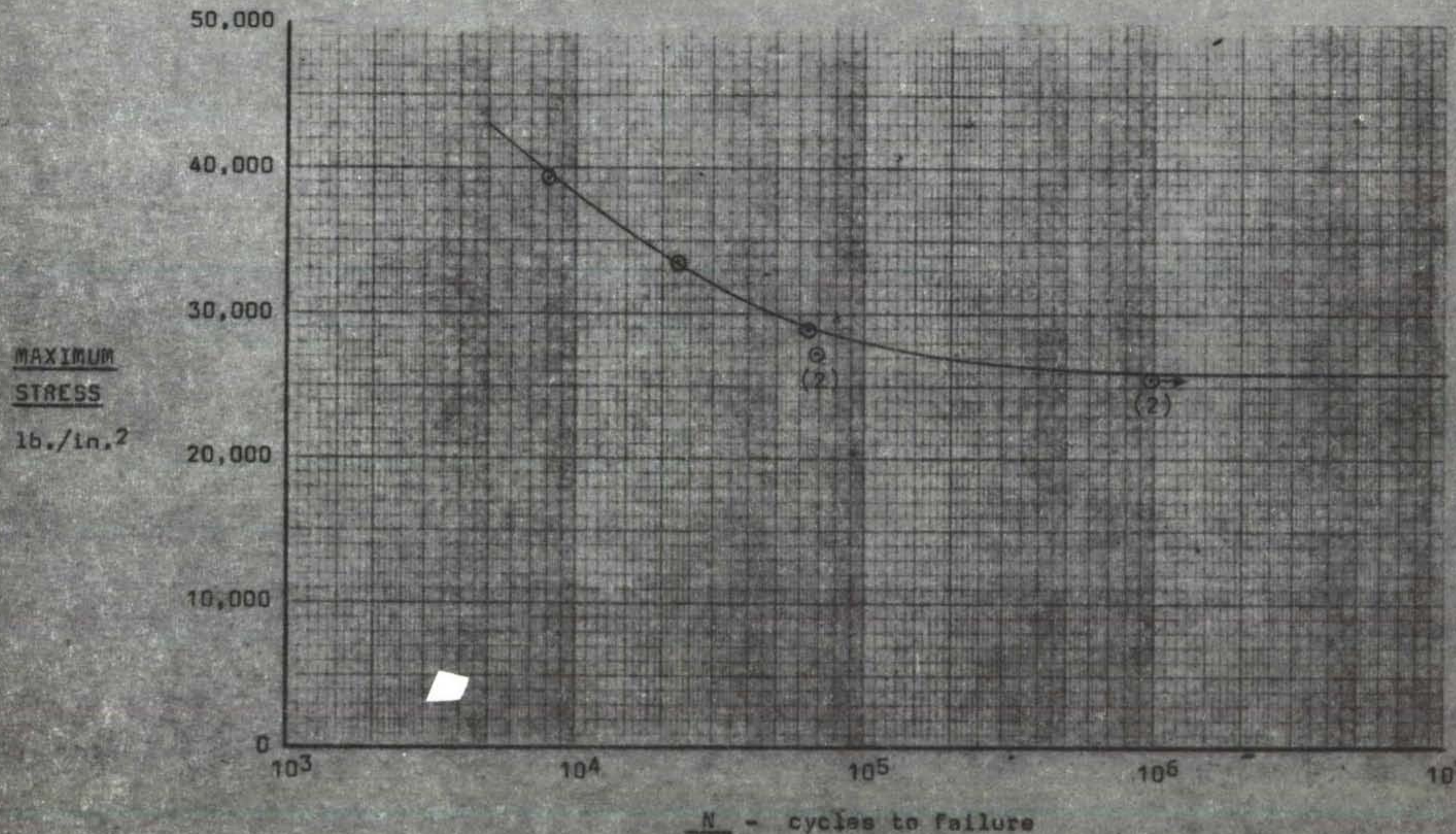

Figure 32. Test No. 2. Open Hole Control. 20,000 psi Mean Stress.

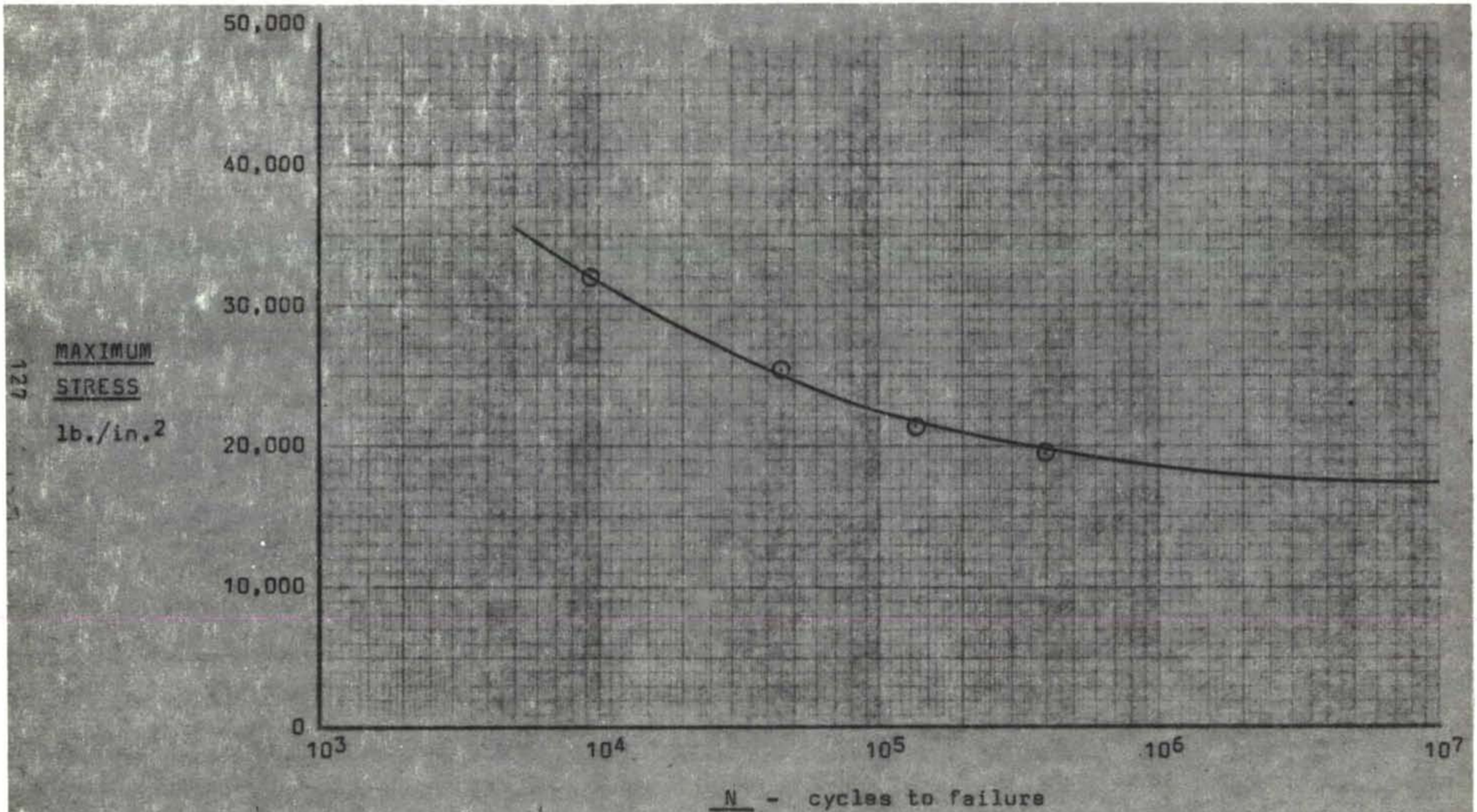

Figure 33. Test No. 1. Open Hole Control. 10,000 psi Mean Stress.

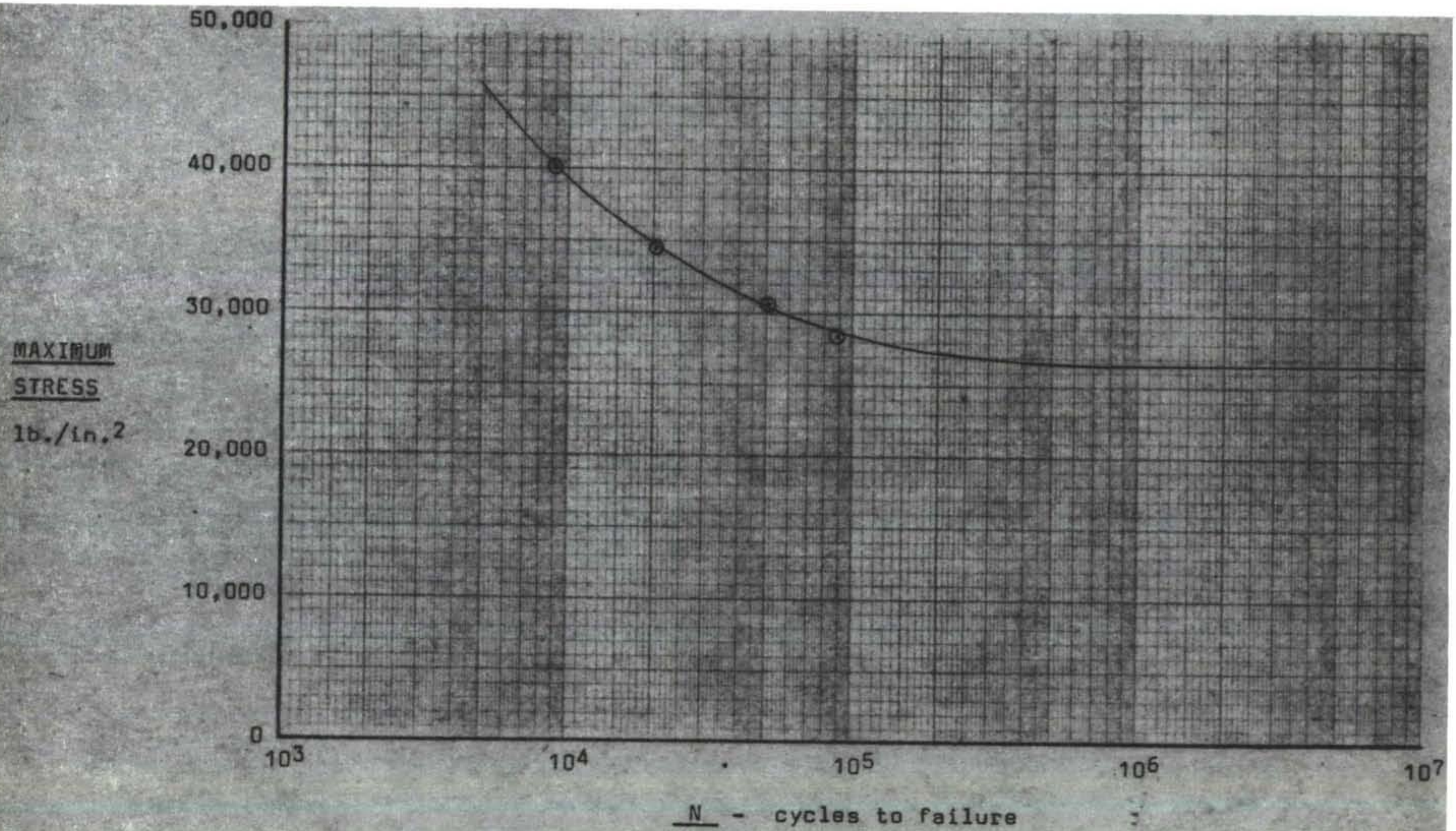

Figure B34. Test No. 3. Open Hole Control. .125 Inch Thickness.

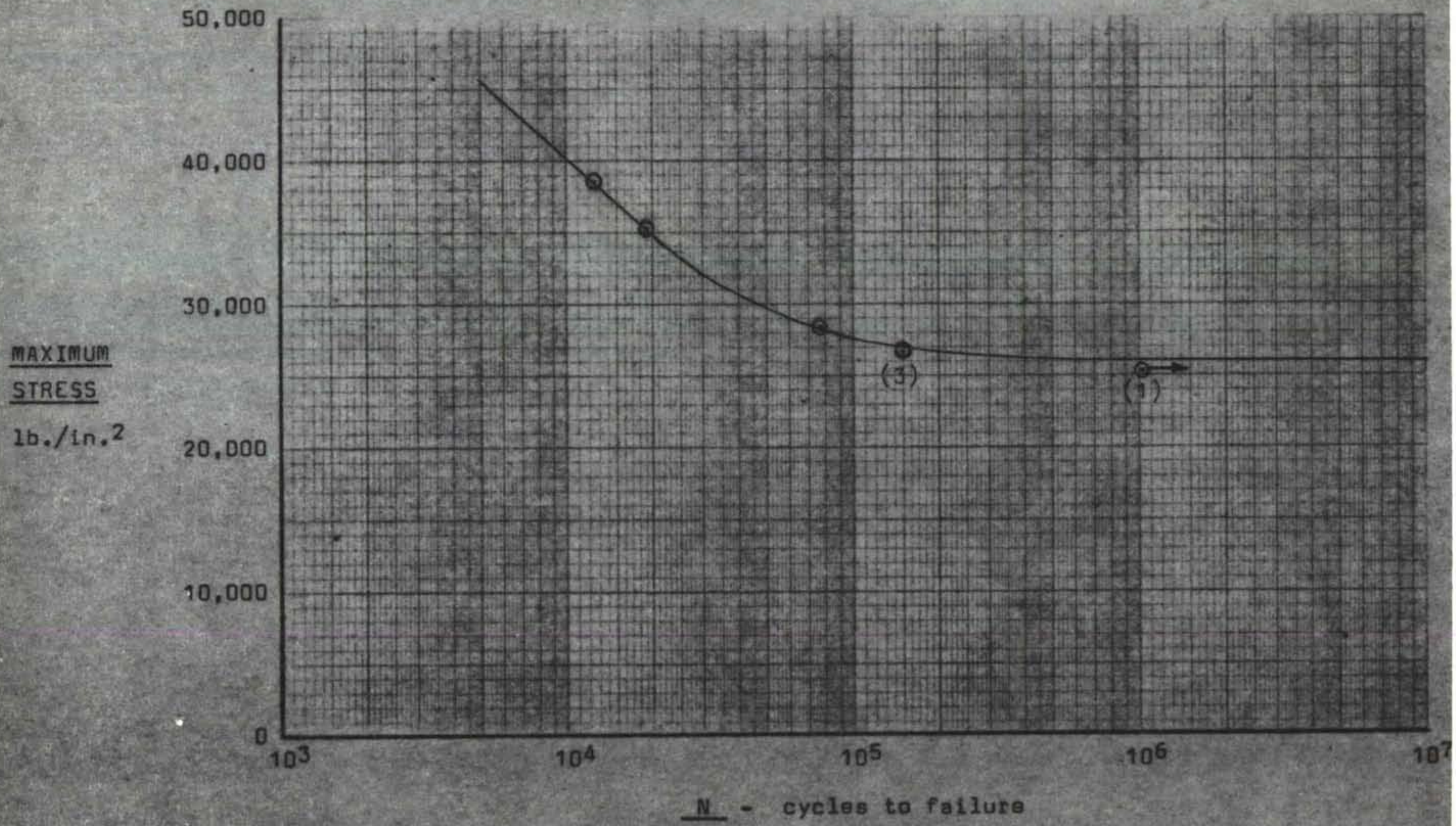

Figure 35. Test No. 4. Open Hole Control. Net Section, 90%.

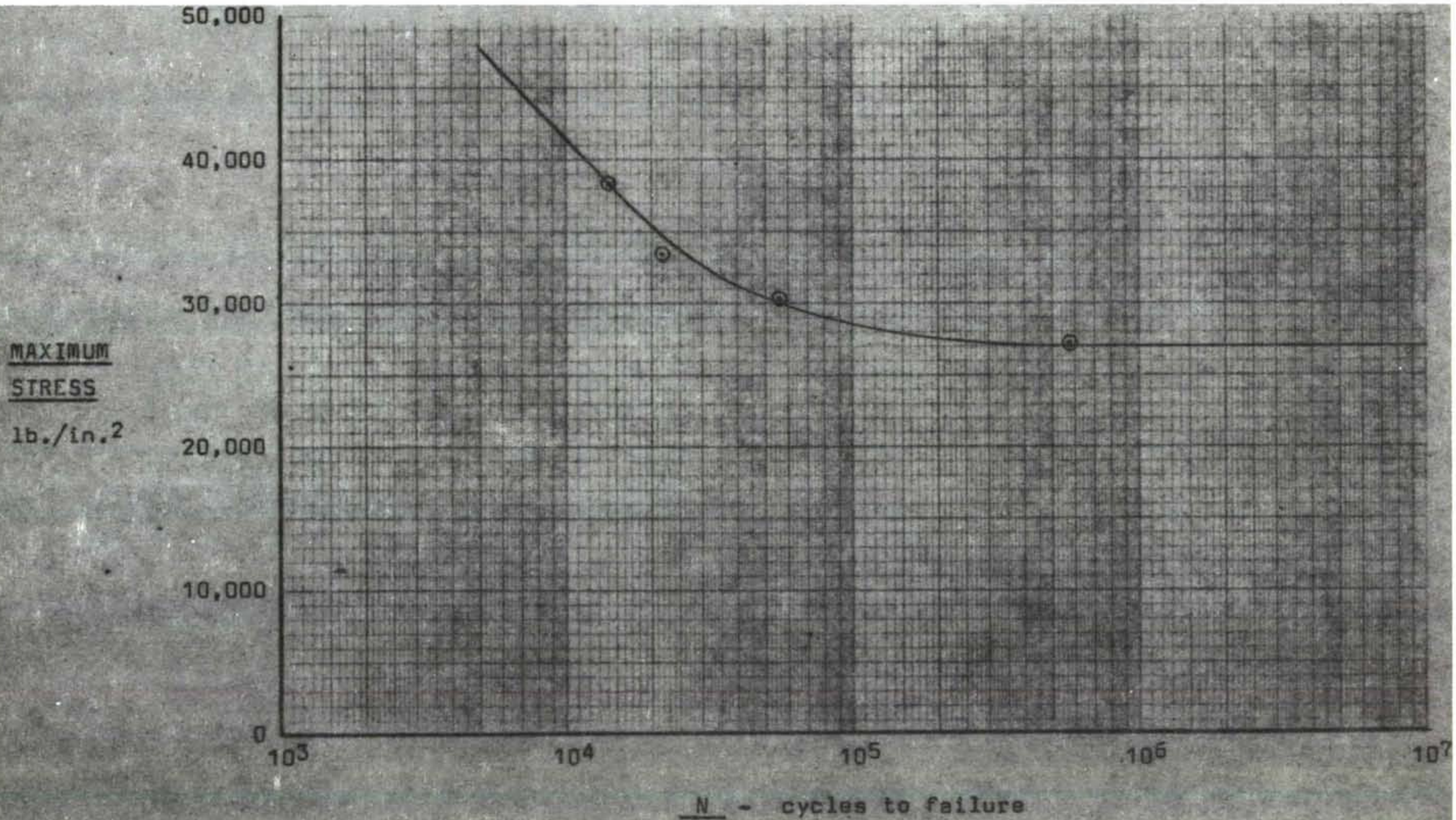

Figure 36. Test No. 5. Open Hole Control. Net Section, 75%.

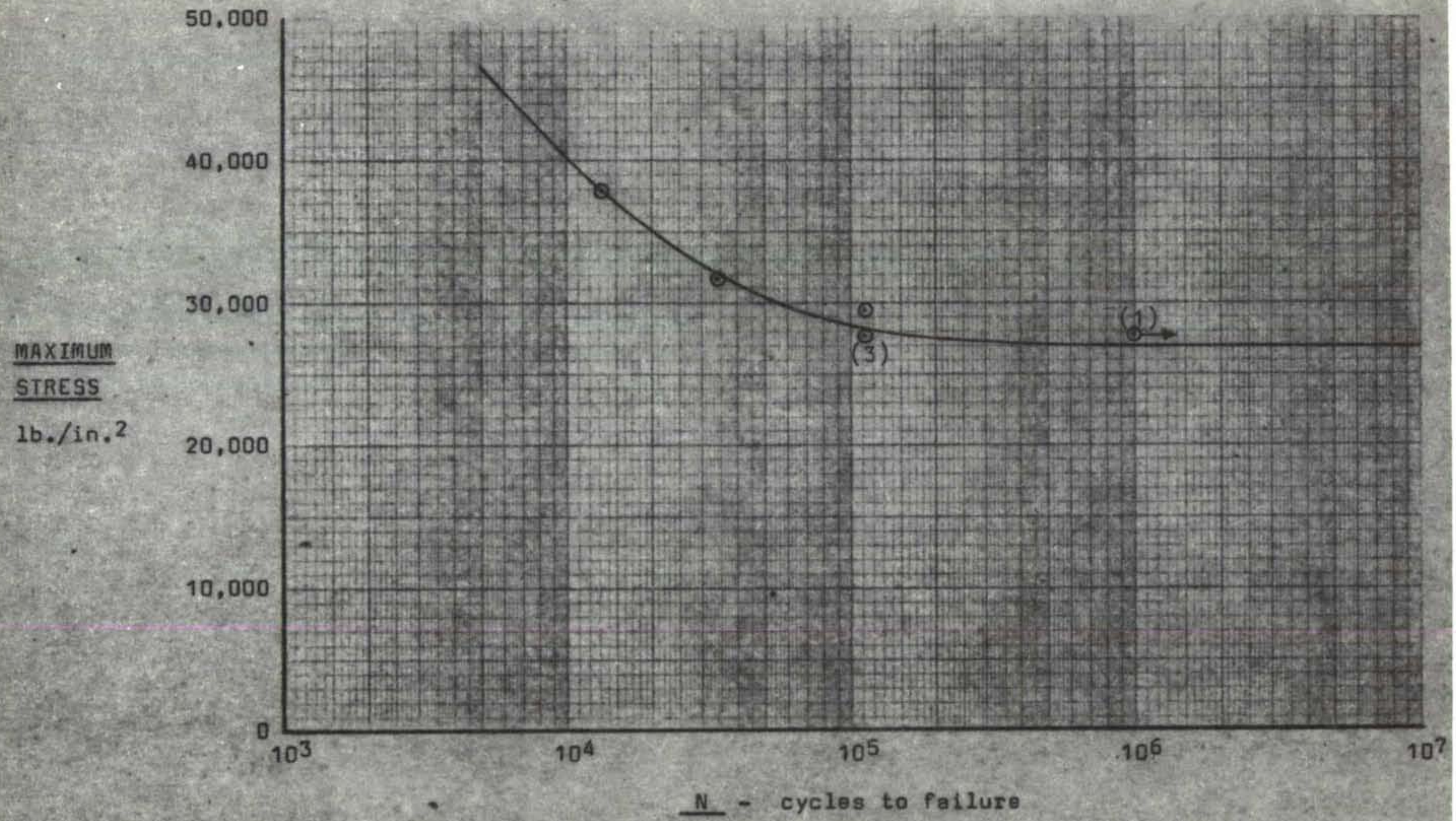

Figure 37. Test No. 6. Open Hole Control. Net Section, 65%.

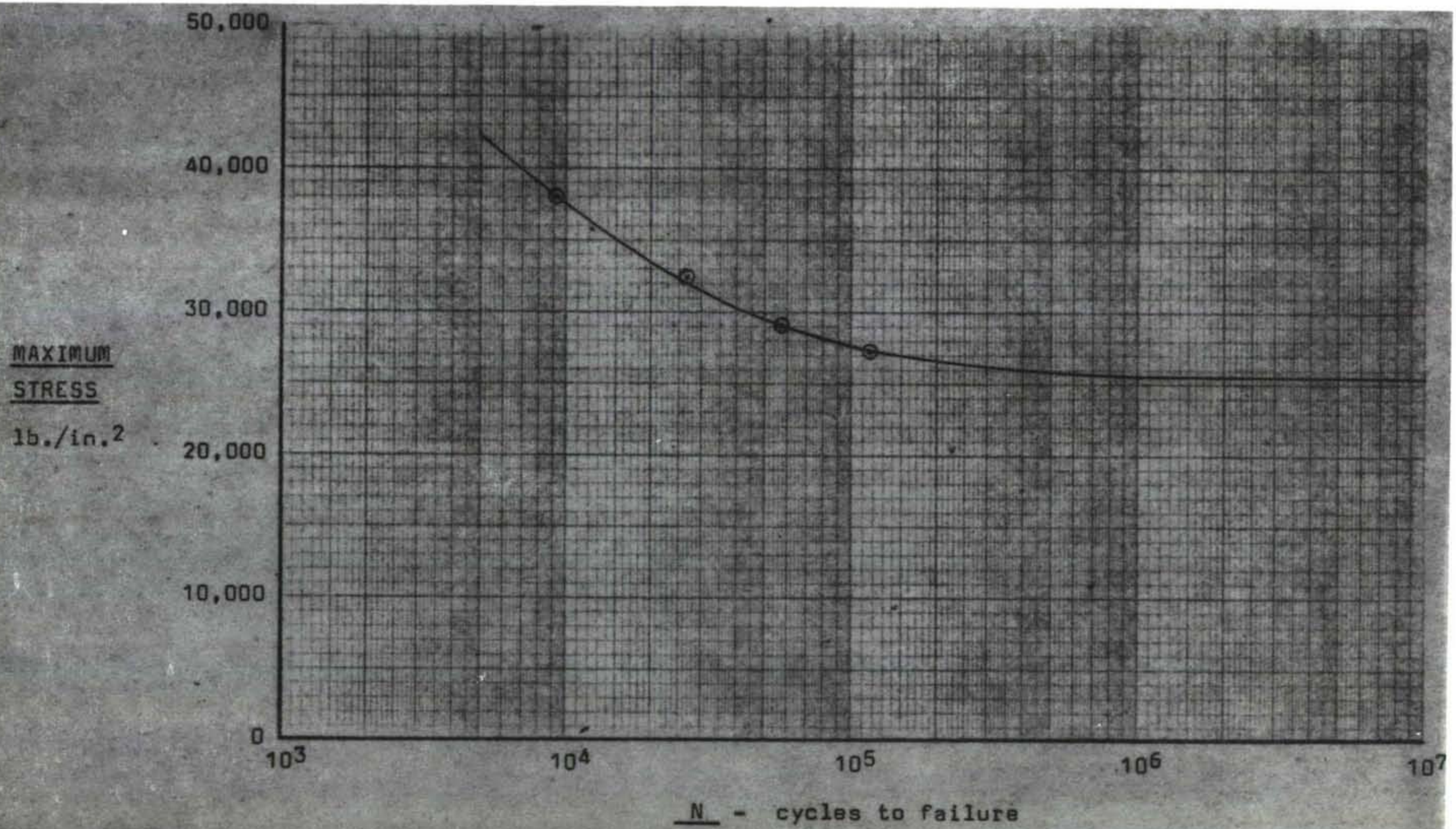

Figure 38. Test No. 9. Open Hole Control. Edge Distance Ratio, 2.5

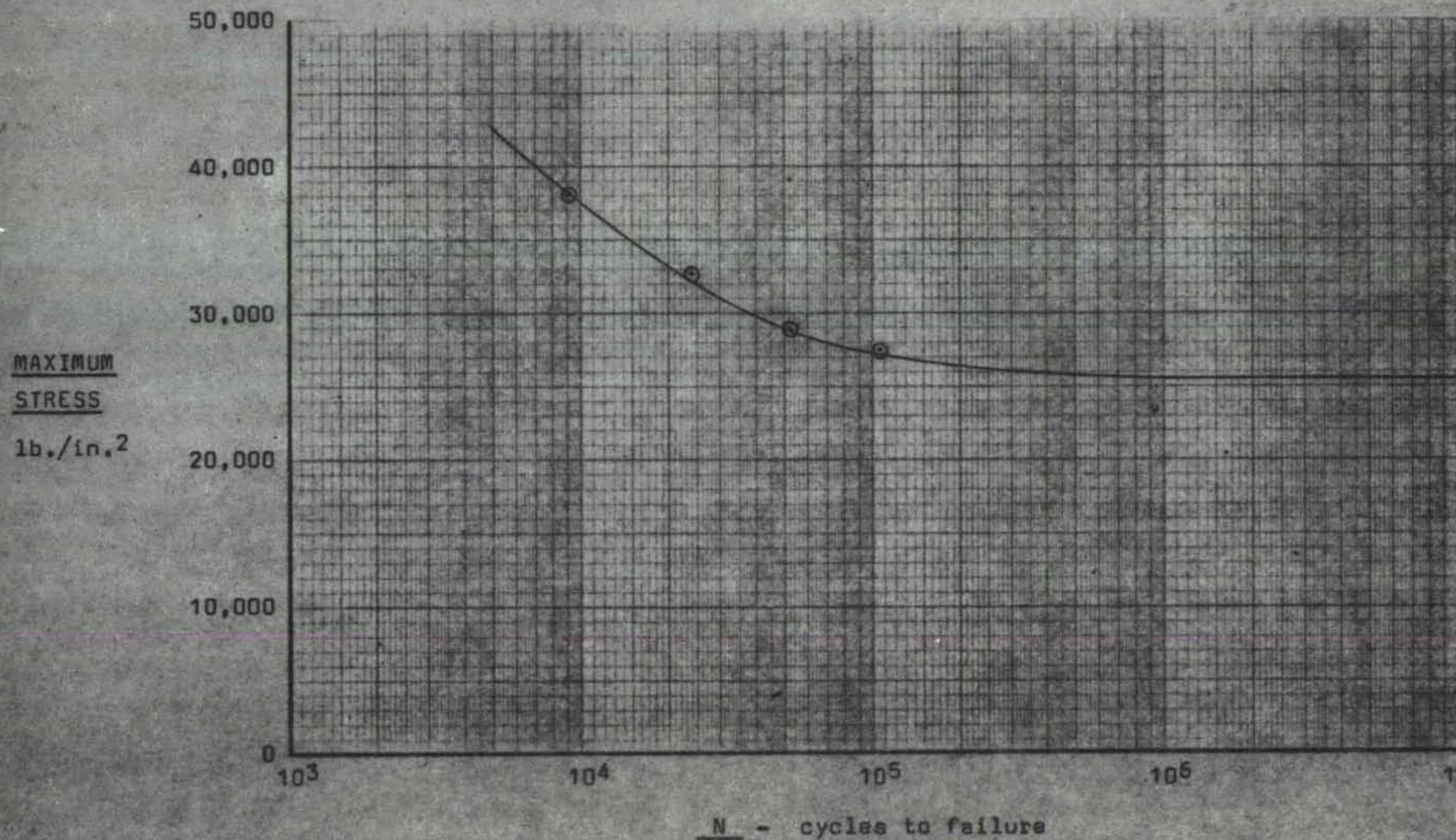

Figure 39. Test No. 8. Open Hole Control. Edge Distance Ratio, 2.0

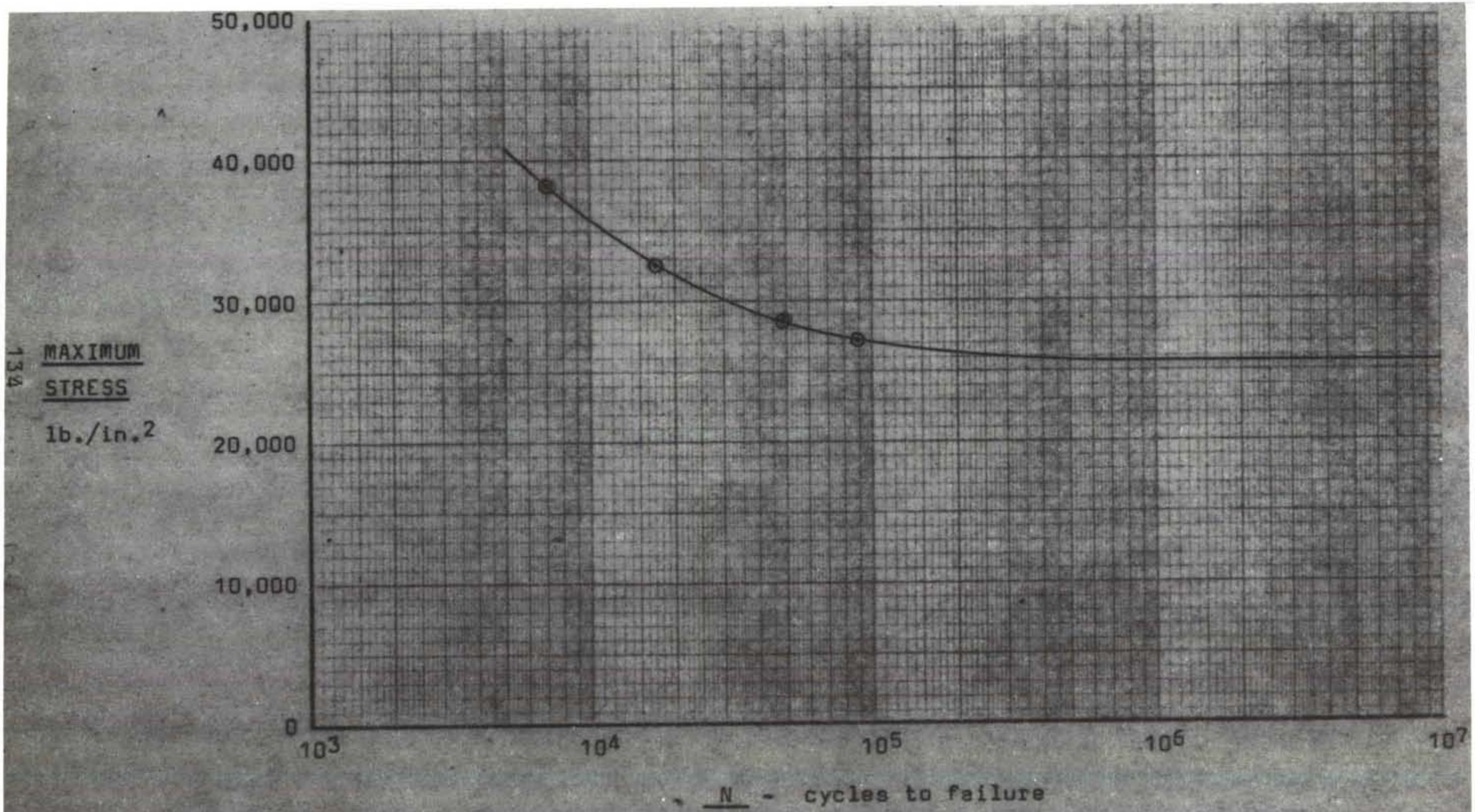

Figure 40. Test No. 7. Open Hole Control. Edge Distance Ratio, 1.5.

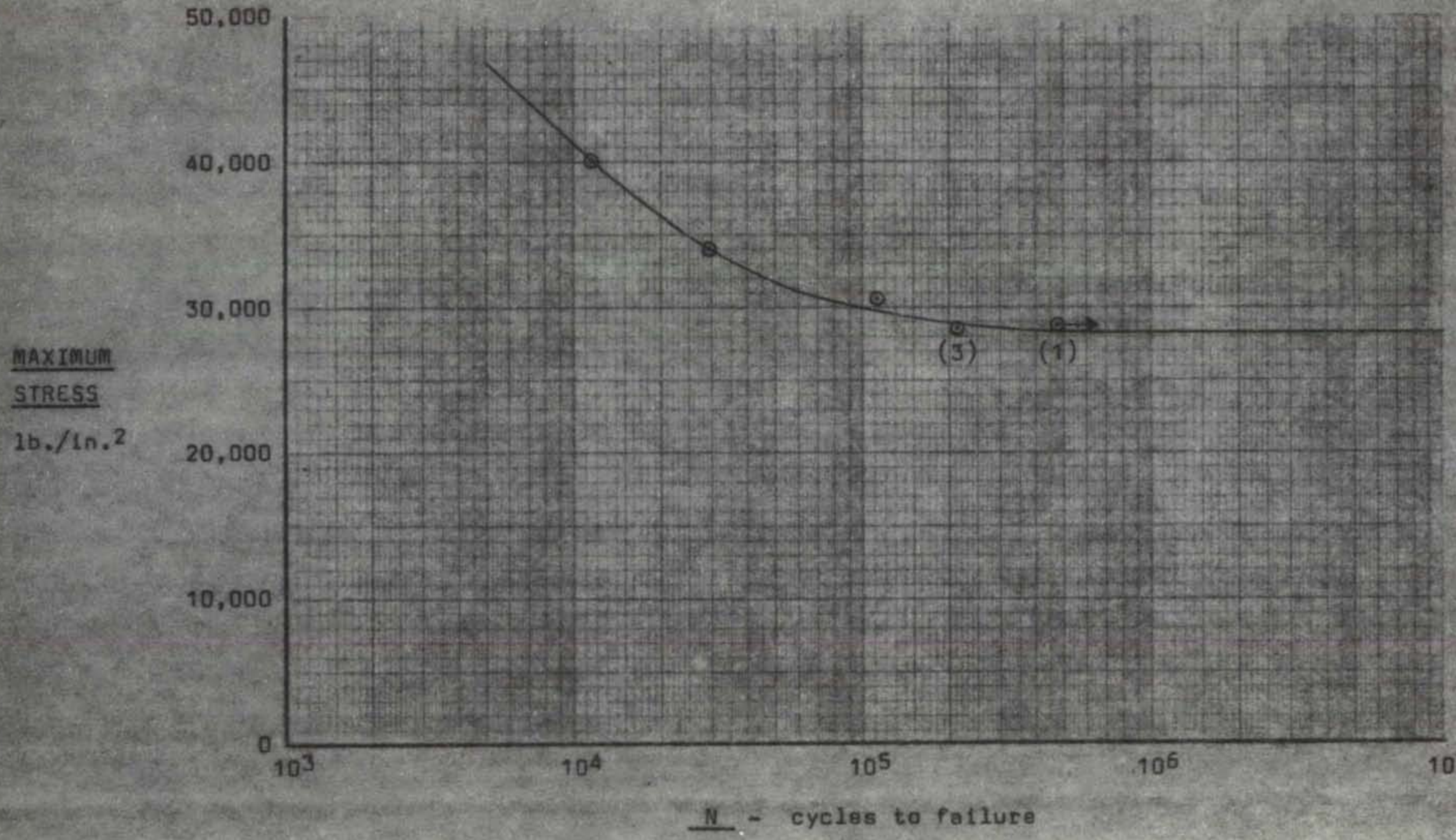

Figure 41. Test No. 34. Bare Material Control.

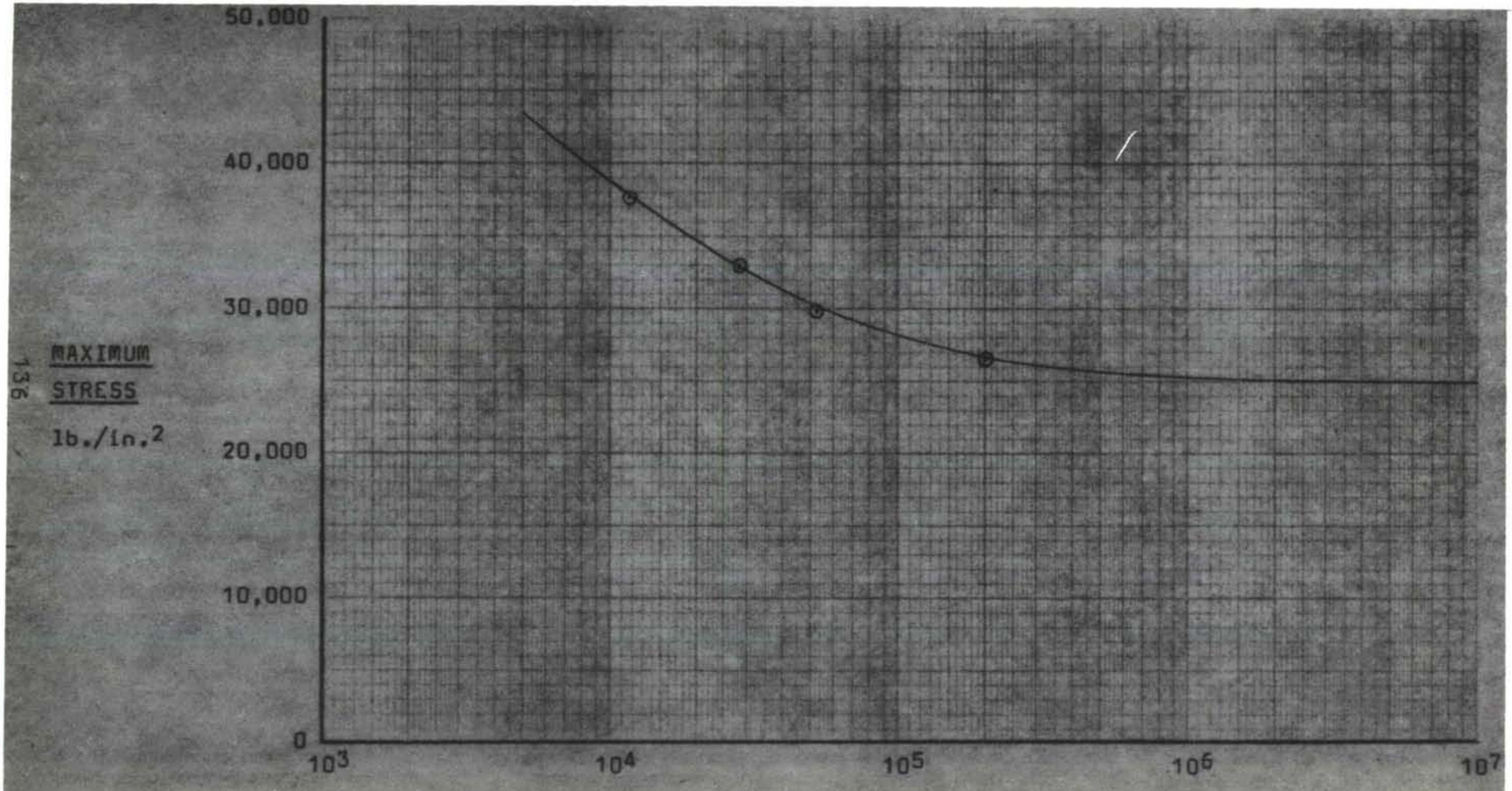

Figure 42. Test No. 38. Control, Open Hole, Surface Covered with Grease. Hole Diameter, .312 in.

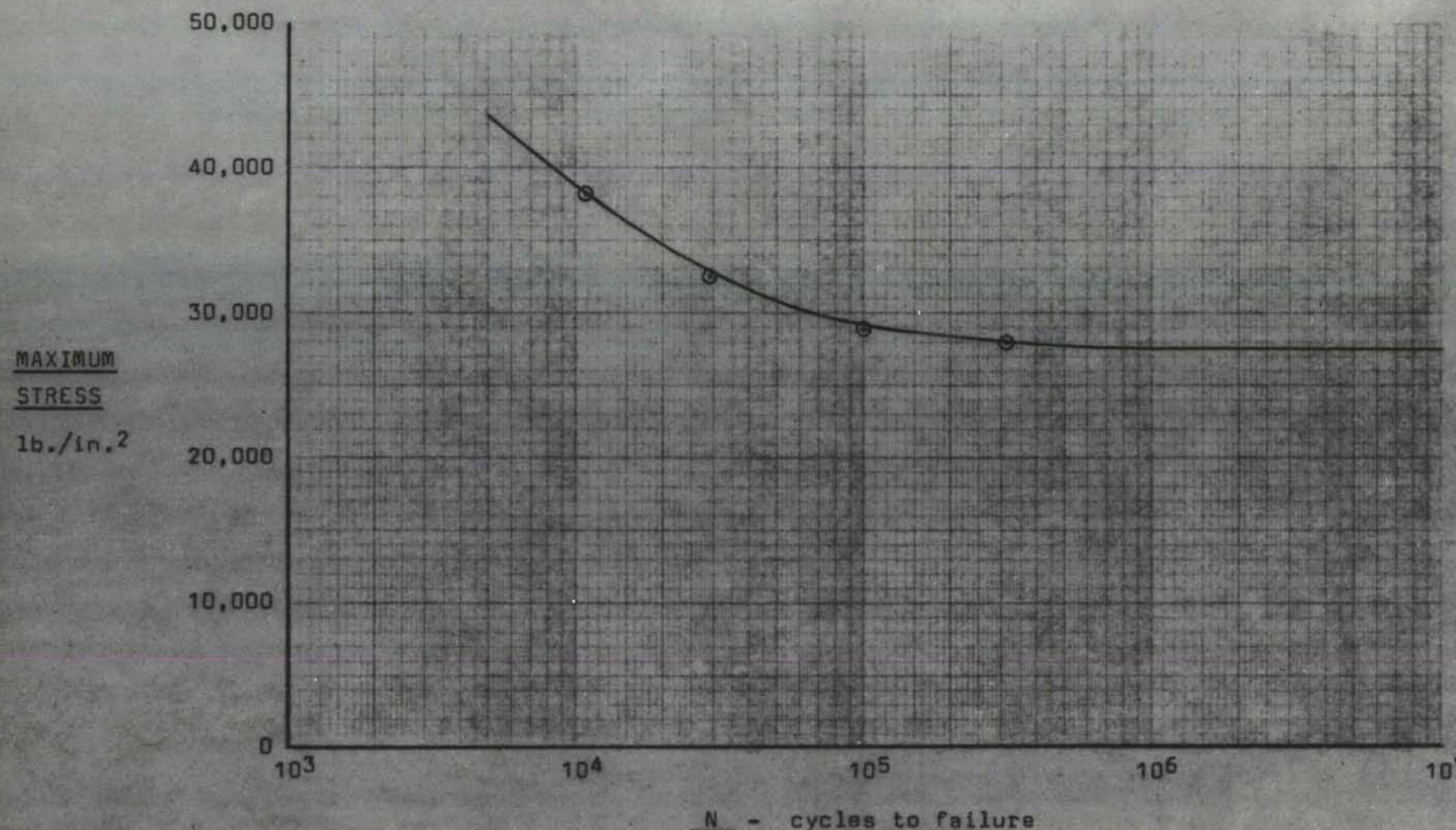

Figure 43. Test No. 14. Open Hole Reaming. Hole Diameter Increase, .03 in., Ream at 25-66% Life.

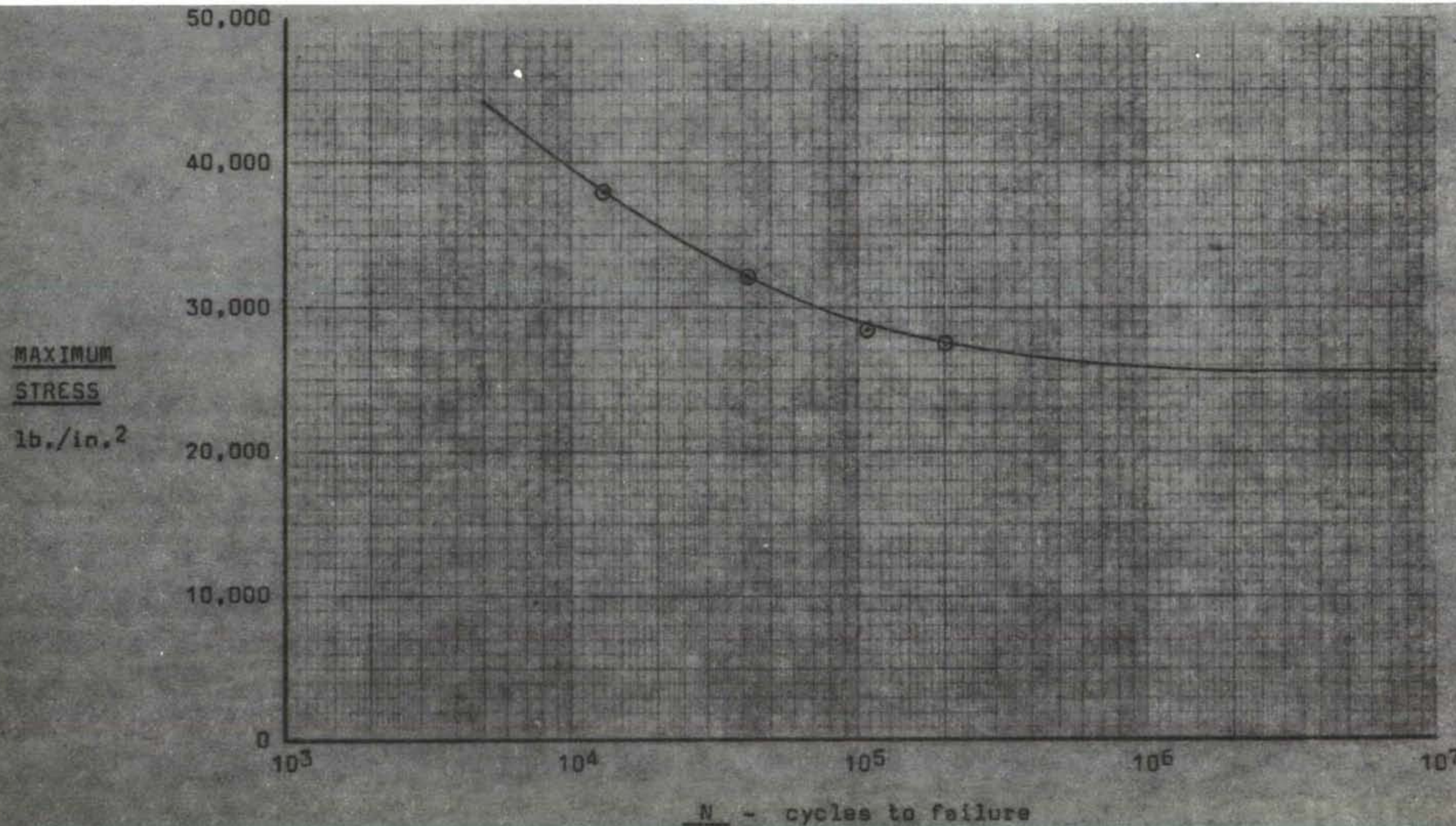

Figure 44. Test No. 15. Open Hole Reaming. Hole Diameter Increase, .03 in., Ream at 25-66% Life.

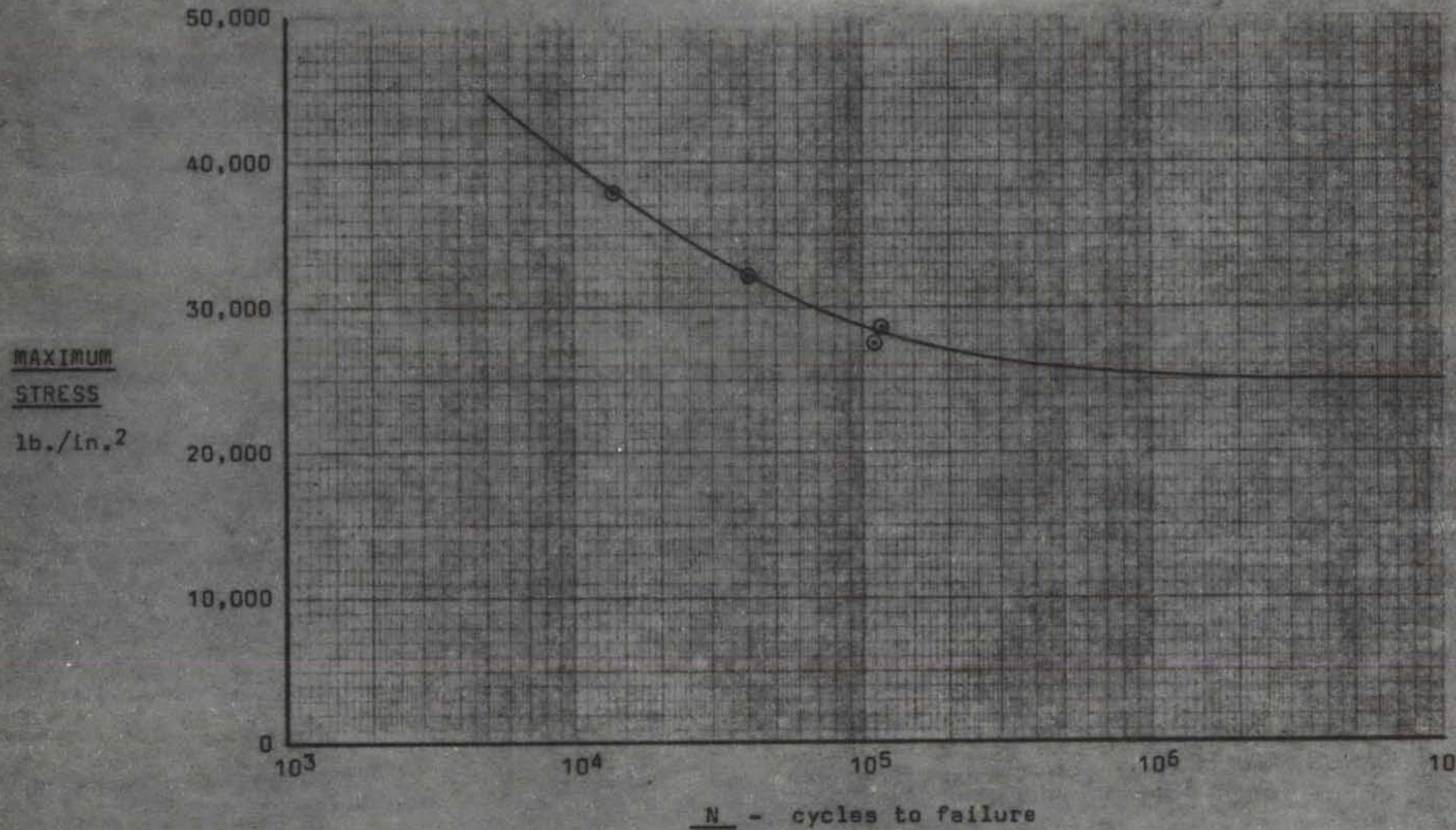

Figure 45. Test No. 16. Open Hole Reaming. Hole Diameter Increase, .03 in., Ream at 25-66% Life.

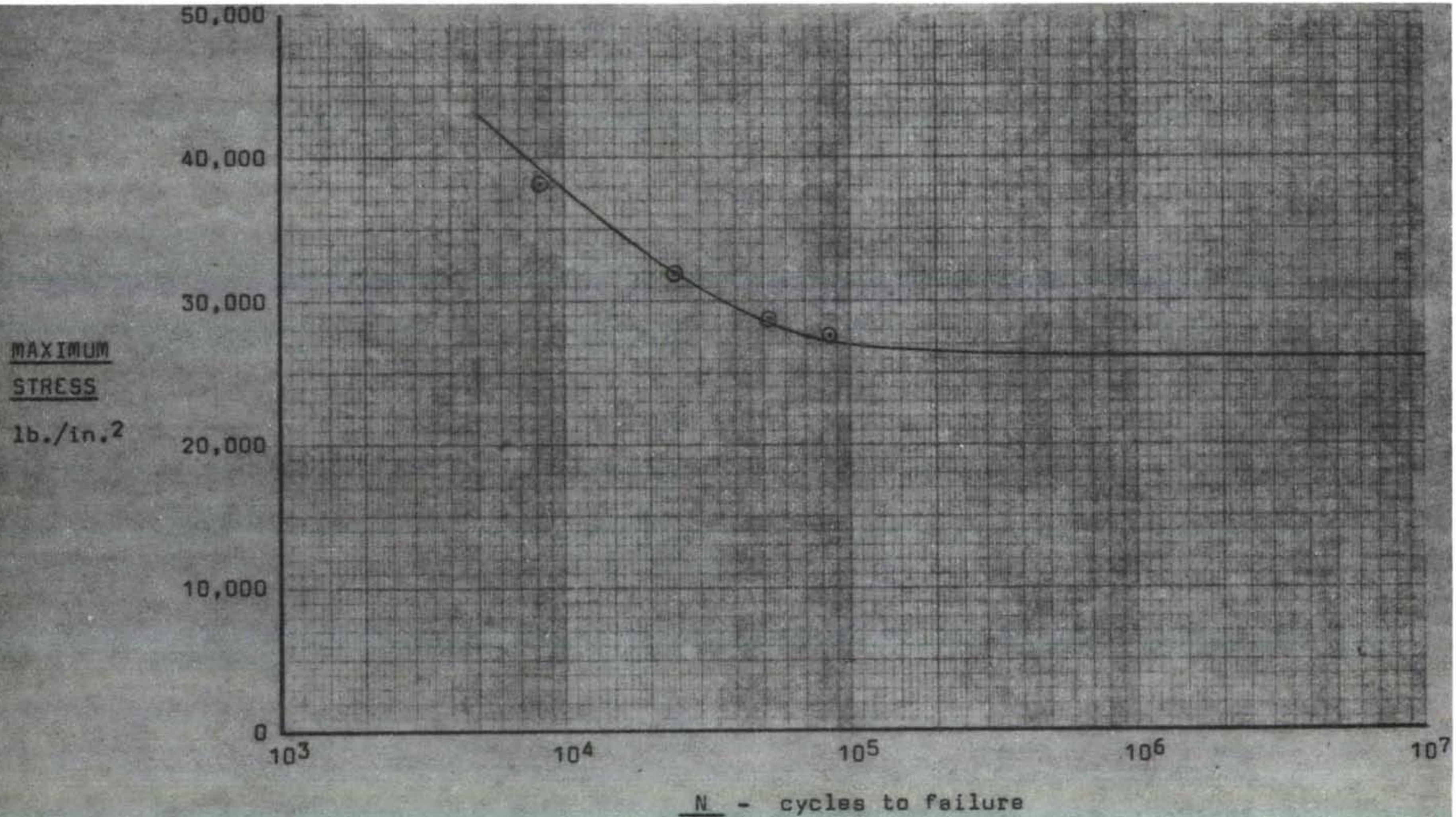

Figure 46. Test No. 17. Open Hole Reaming. Hole Diameter Increase, .03 in., Ream at 100% Life (Initial Crack).

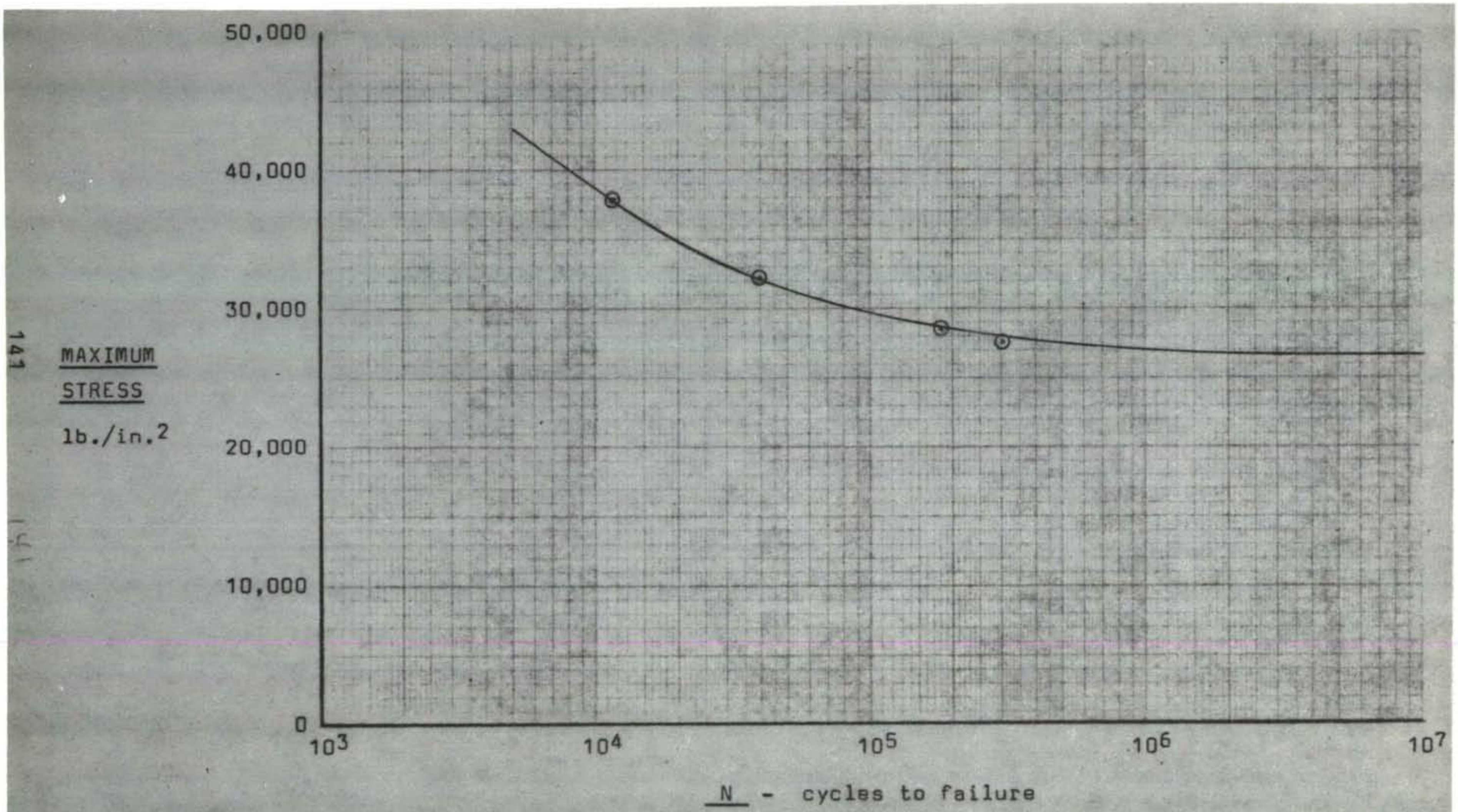

Figure 47. Test No. 18. Open Hole Reaming. Hole Diameter Increase, .06 in., Ream at 25-66% Life.

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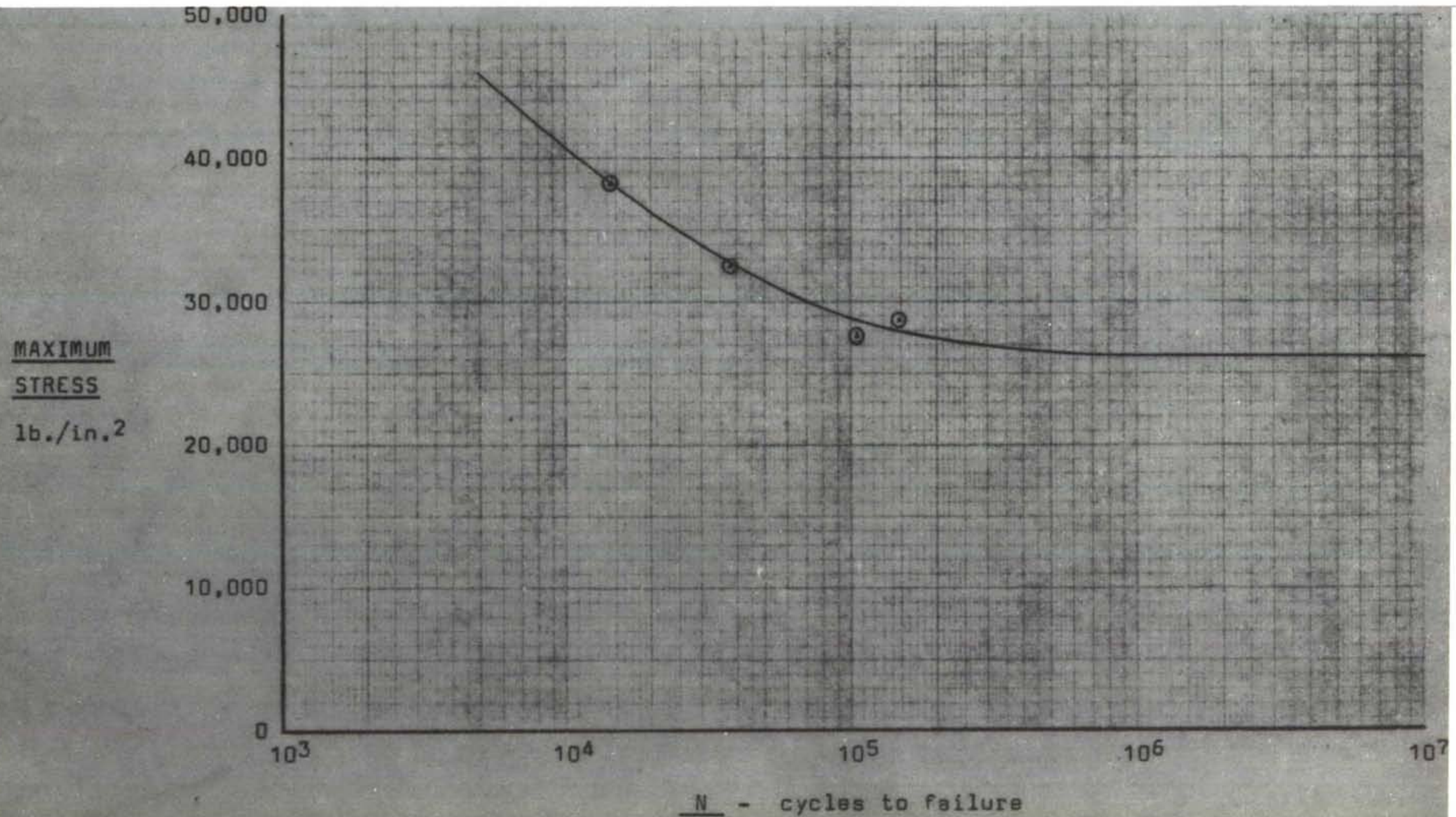

Figure 48. Test No. 19. Open Hole Reaming. Hole Diameter Increase, .06 in., Ream at 25-66% Life.

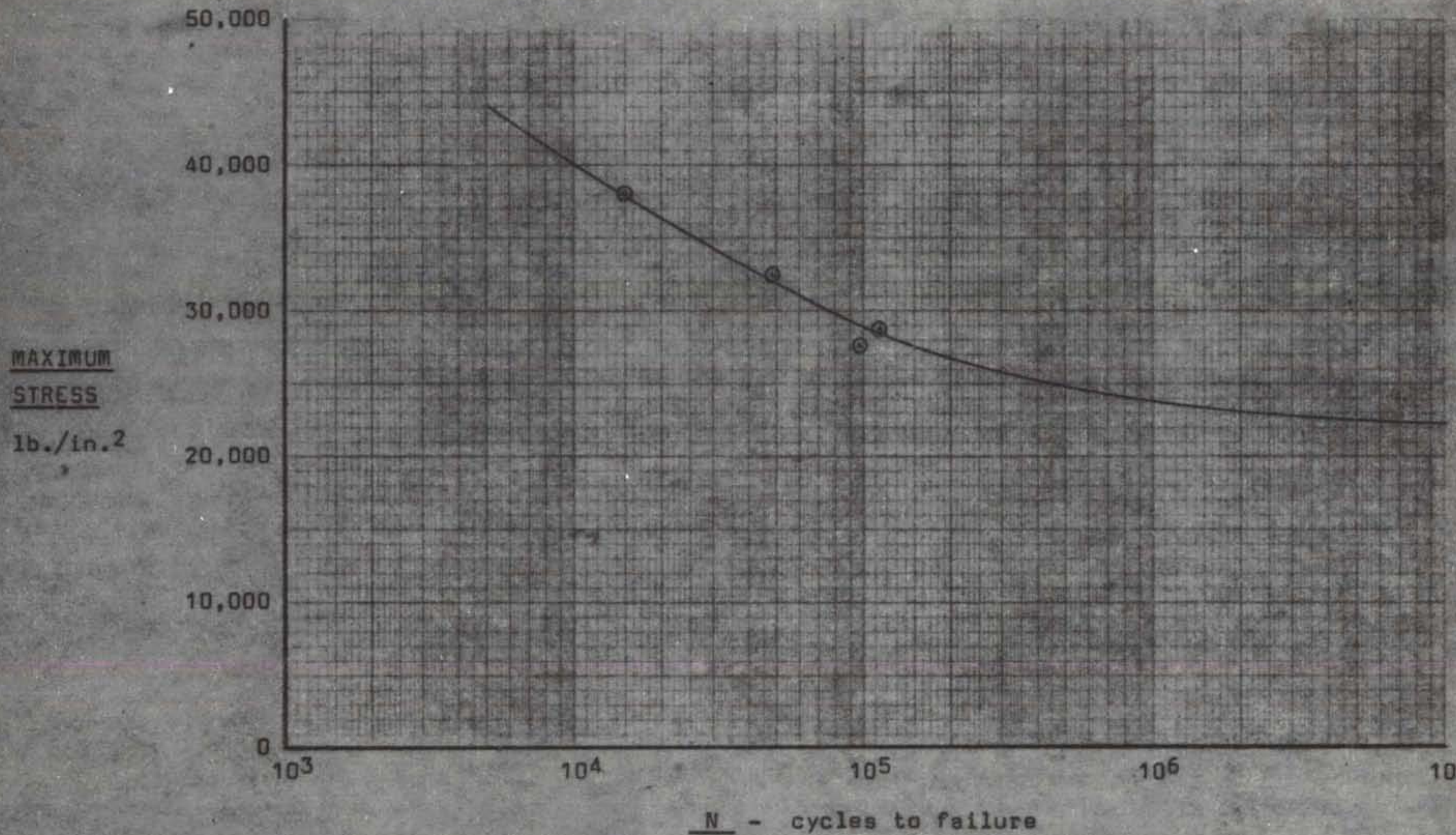

Figure 49. Test No. 20. Open Hole Reaming. Hole Diameter Increase, .06 in., Ream at 25-66% Life.

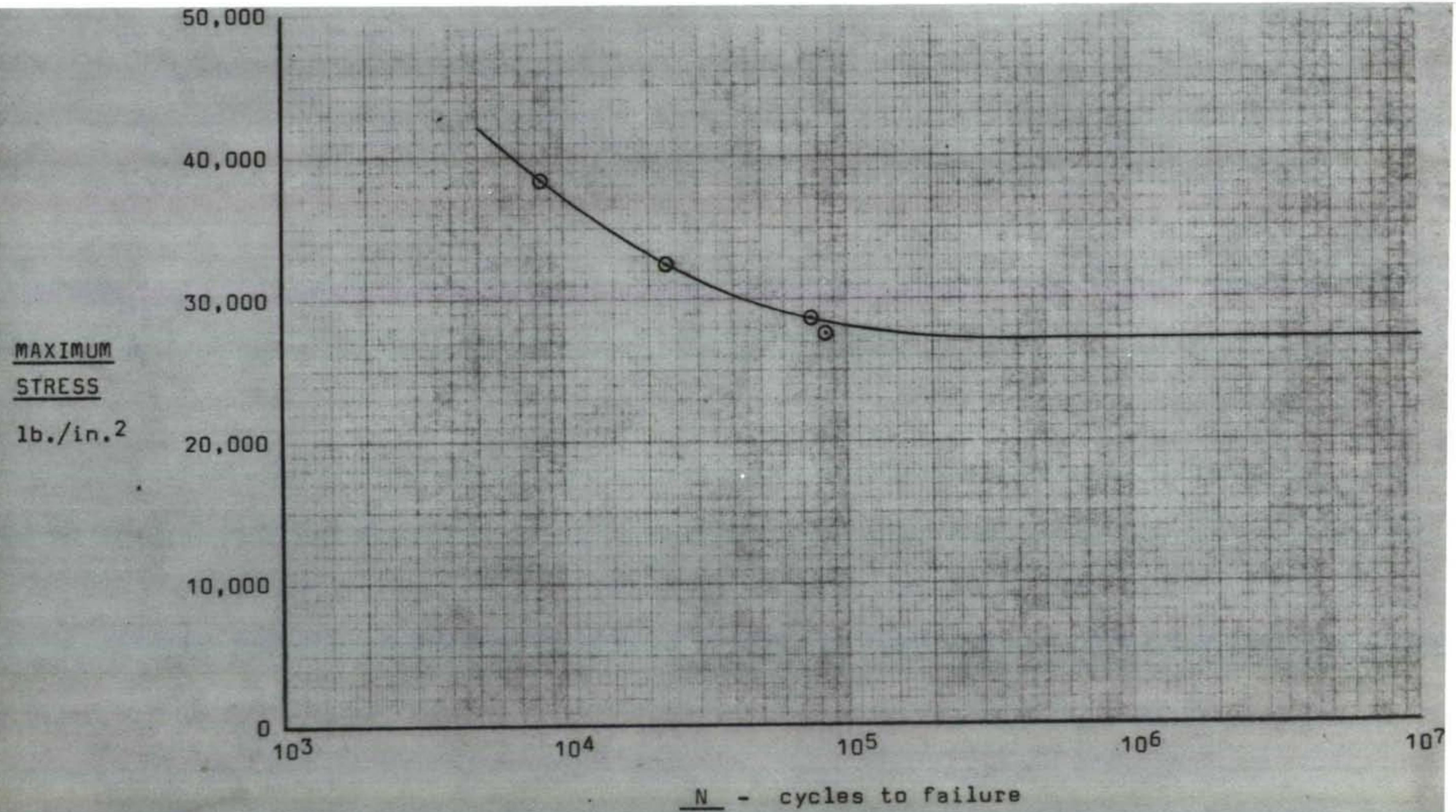

Figure 50. Test No. 21. Open Hole Reaming. Hole Diameter Increase, .06 in., Ream at 100% Life (Initial Crack).

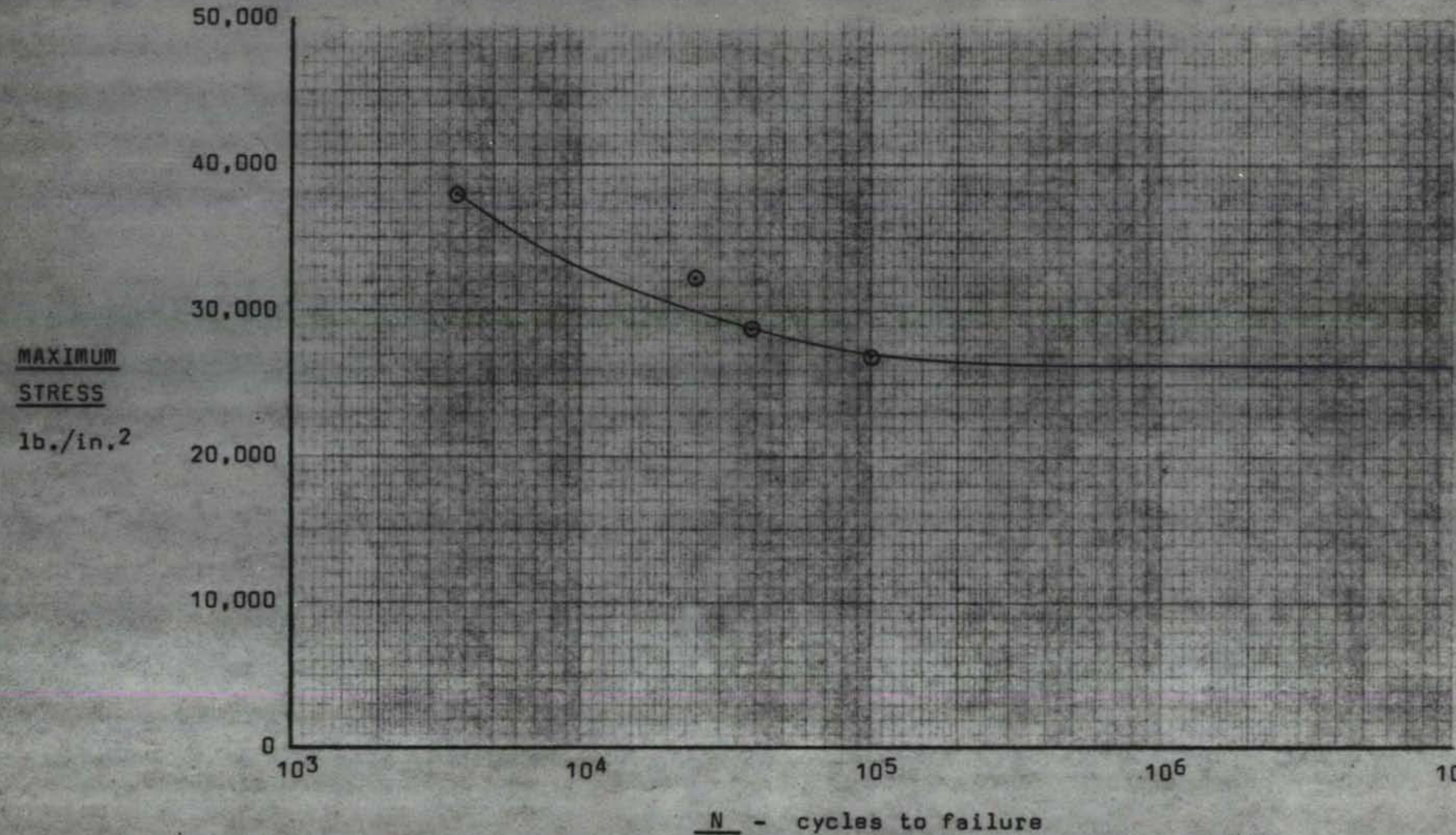

Figure 51. Test No. 50. Short Edge Distance, Double Ream. Edge Distance Ratio Before Reaming, 2.5. Hole Diameter Increase: .06 in. at 50% Life, .03 in. at 100% Life.

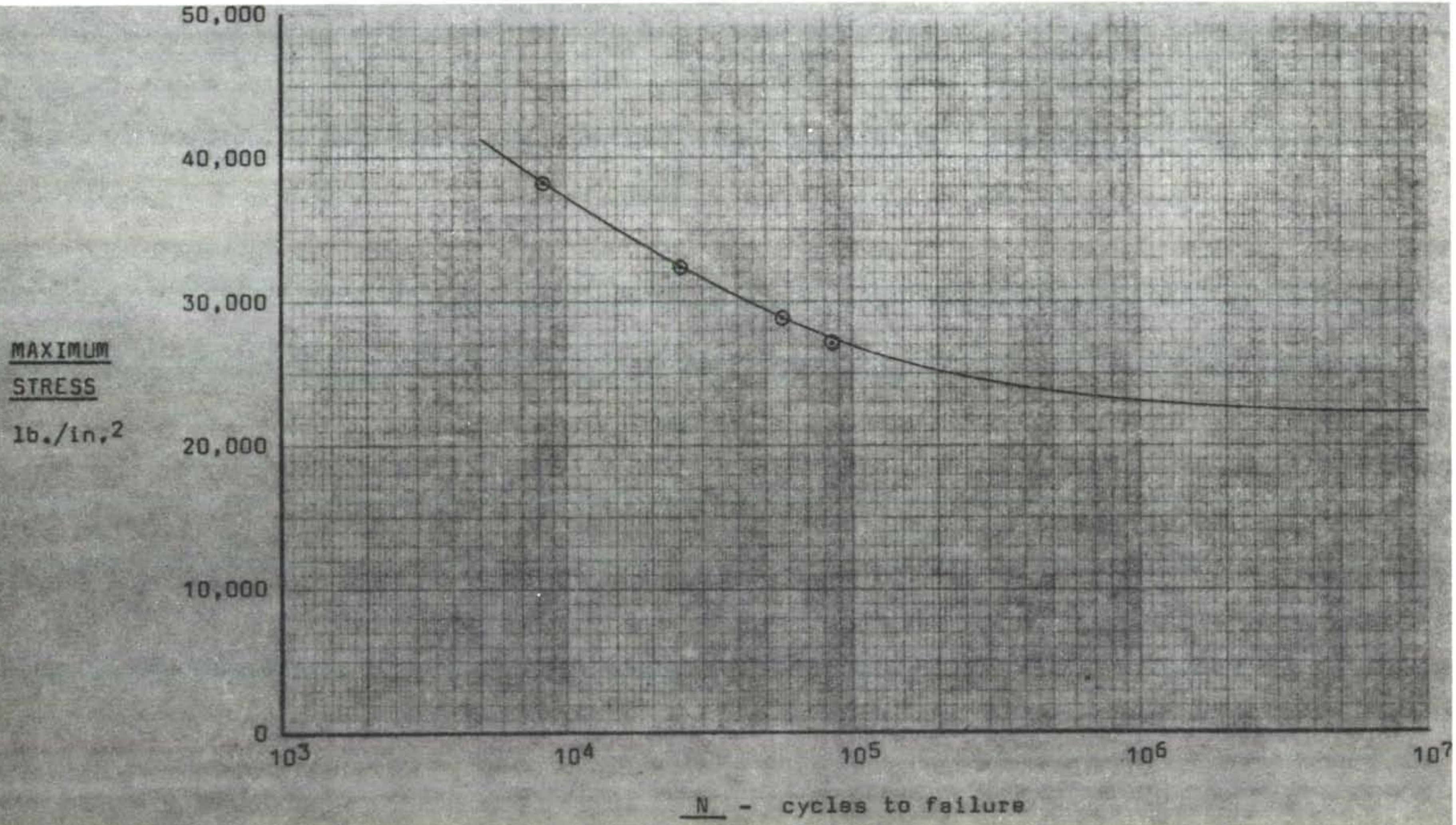

Figure 52. Test No. 40. Short Edge Distance, Double Ream. Edge Distance Ratio Before Reaming, 1.5. Hole Diameter Increase: .06 in. at 50% Life, .03 in. at 100% Life.

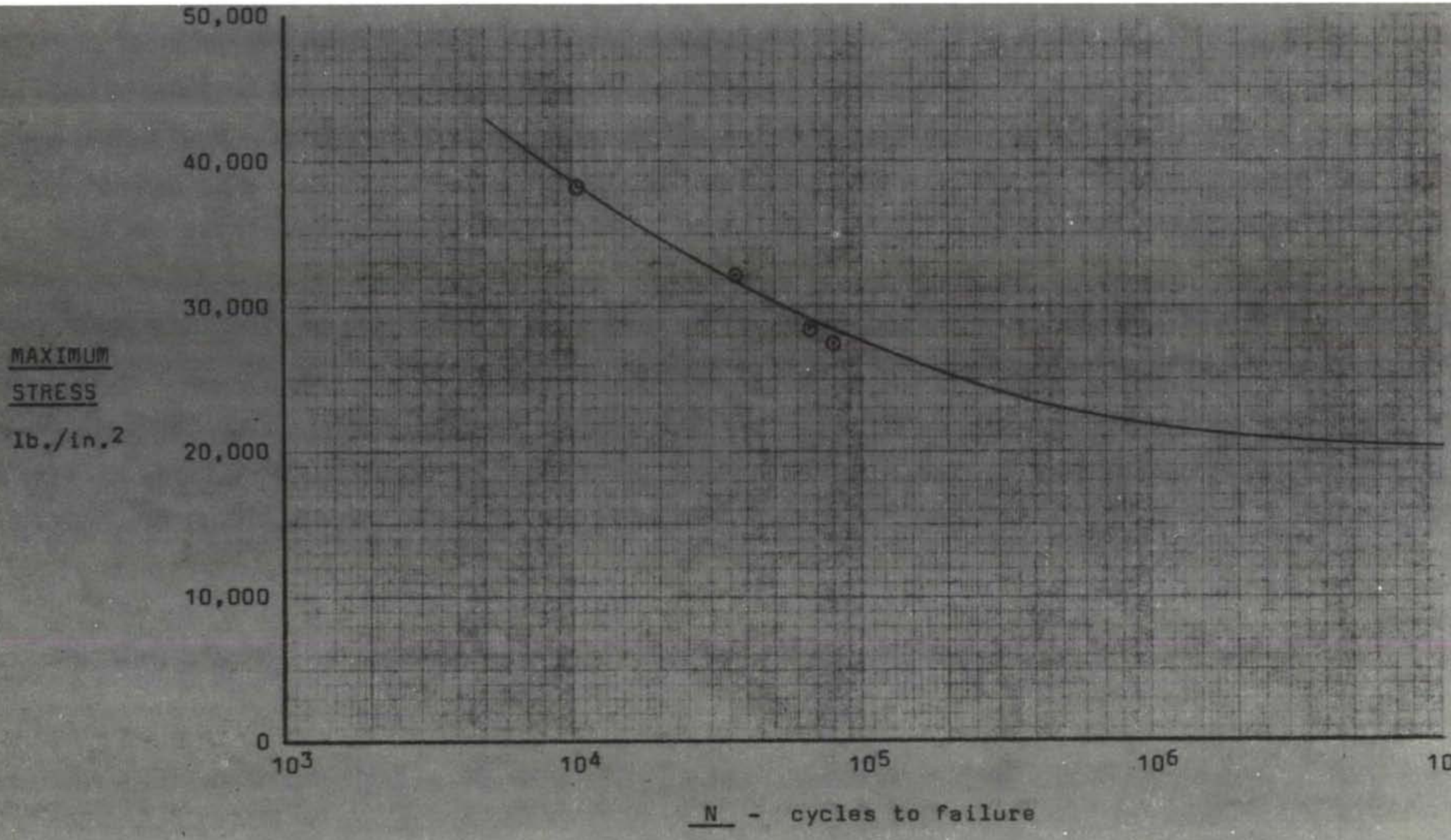

Figure 53.. Test No. 26. Open Hole Peening (.007A).

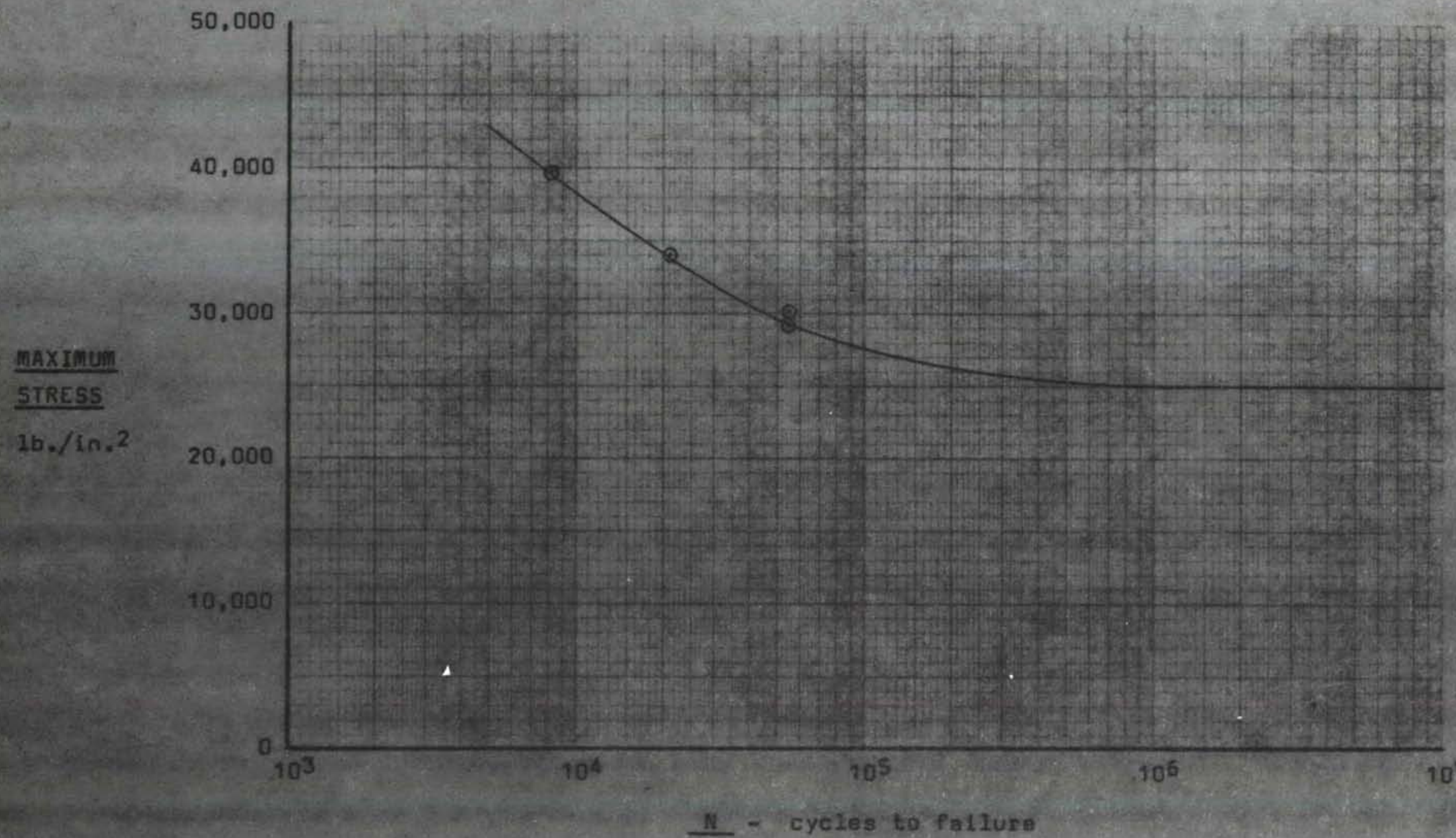

Figure 54. Test No. 35. Open Hole Peening, Bare Material (.007A).

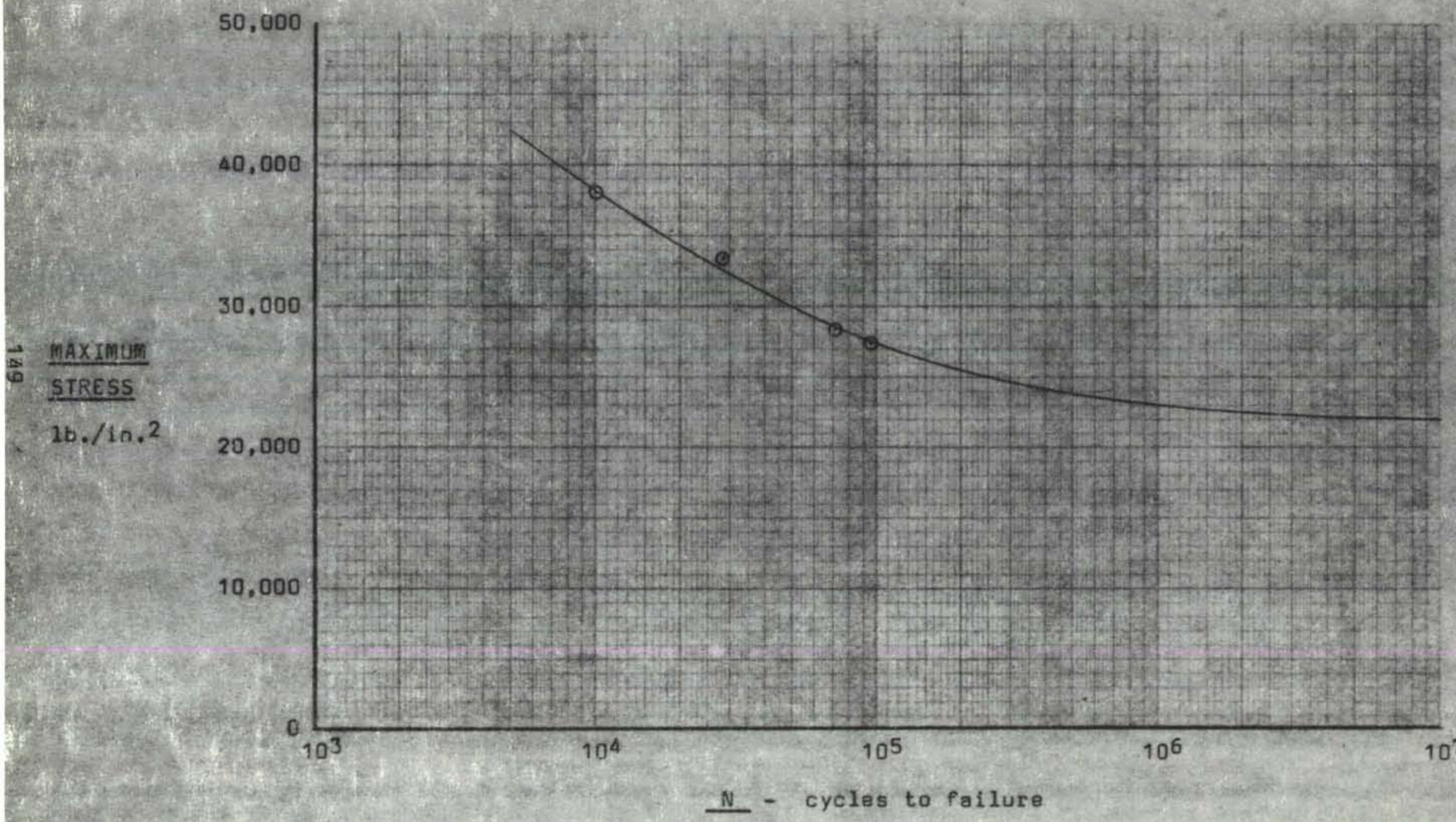

Figure 55. Test No. 37. Open Hole Peening (.007A). .250 in. Hole.

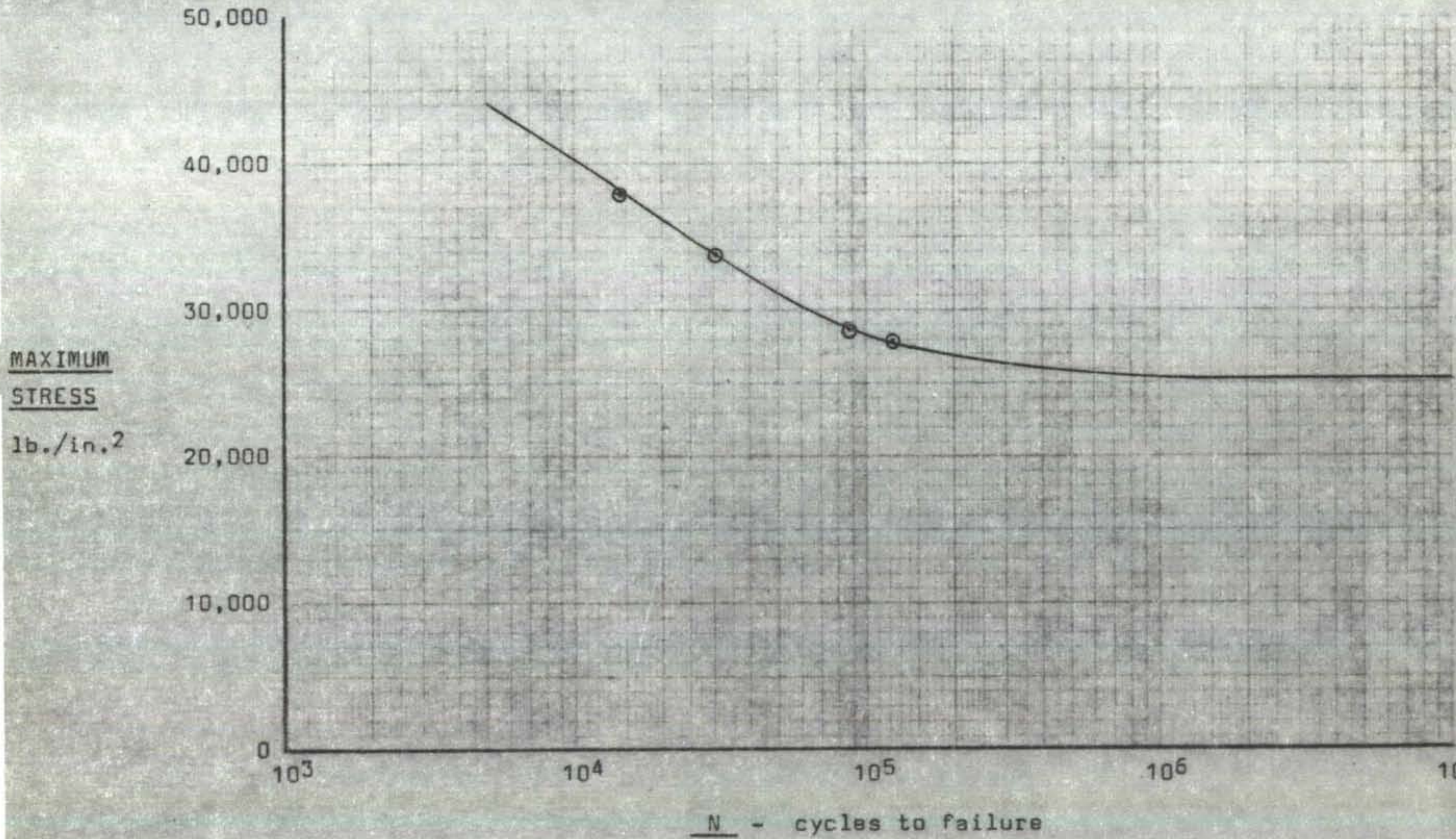

Figure 56.. Test No. 30. Peening on Clad Material. High Intensity Peening, Outside of Hole. (Steel Shot, .015A).

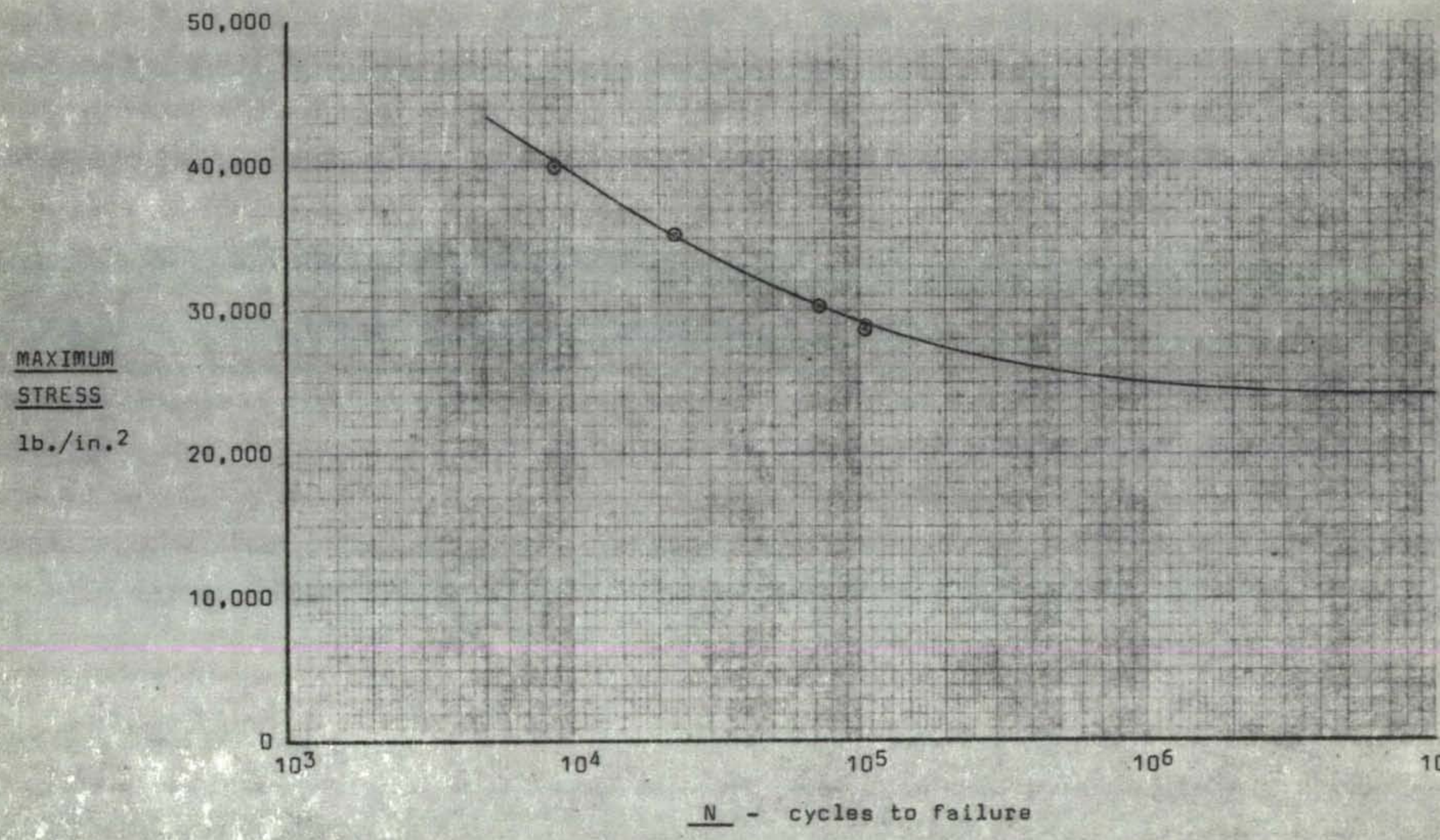

Figure 57. Test No. 29. Peening on Bare Material, .250 in. Hole. High Intensity Peening, Outside of Hole. (Steel Shot, .015A).

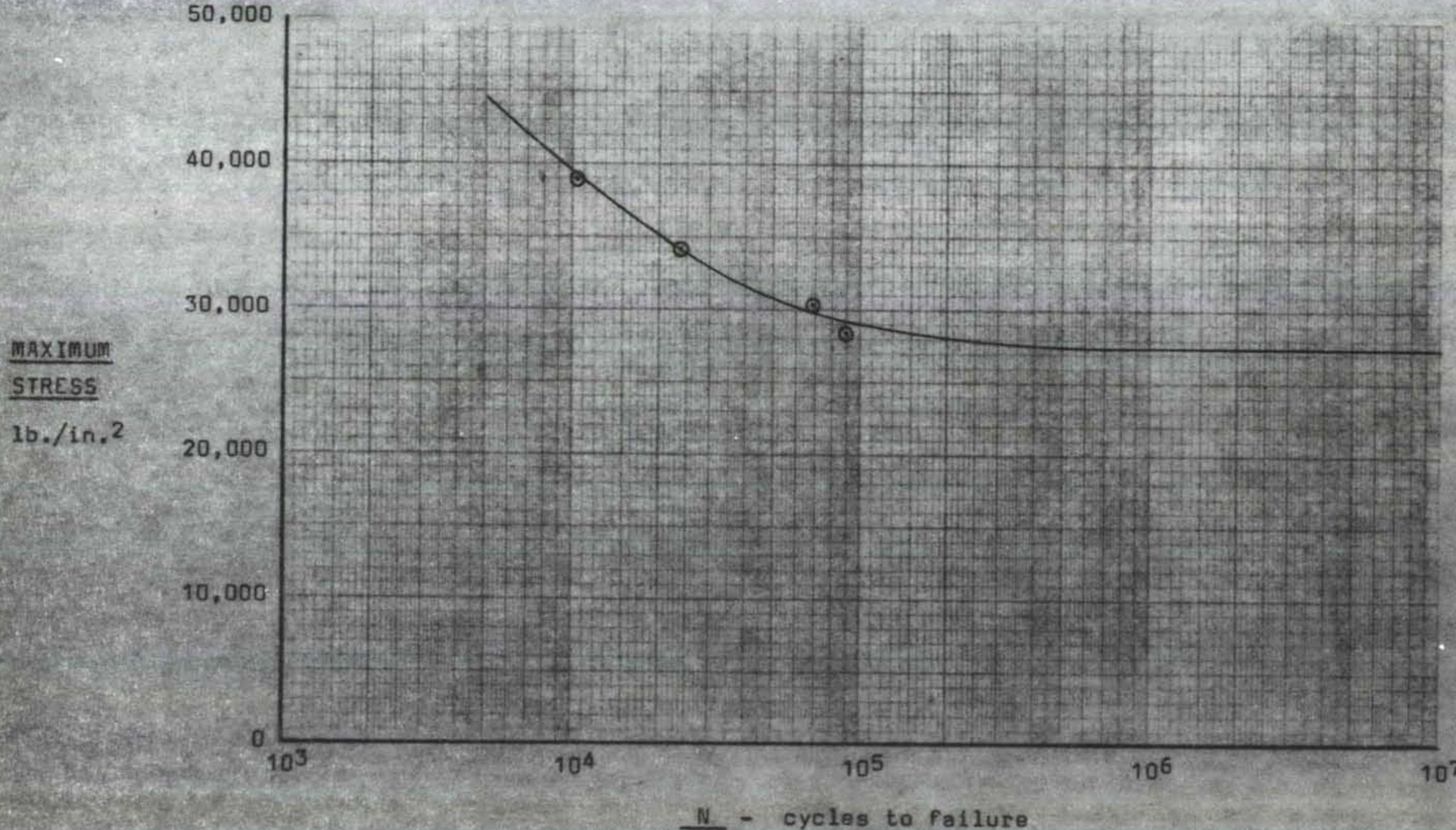

Figure 58. Test No. 36. Open Hole, High Intensity Peening, Outside of Hole (.015A). Specimen Thickness, .125 in.

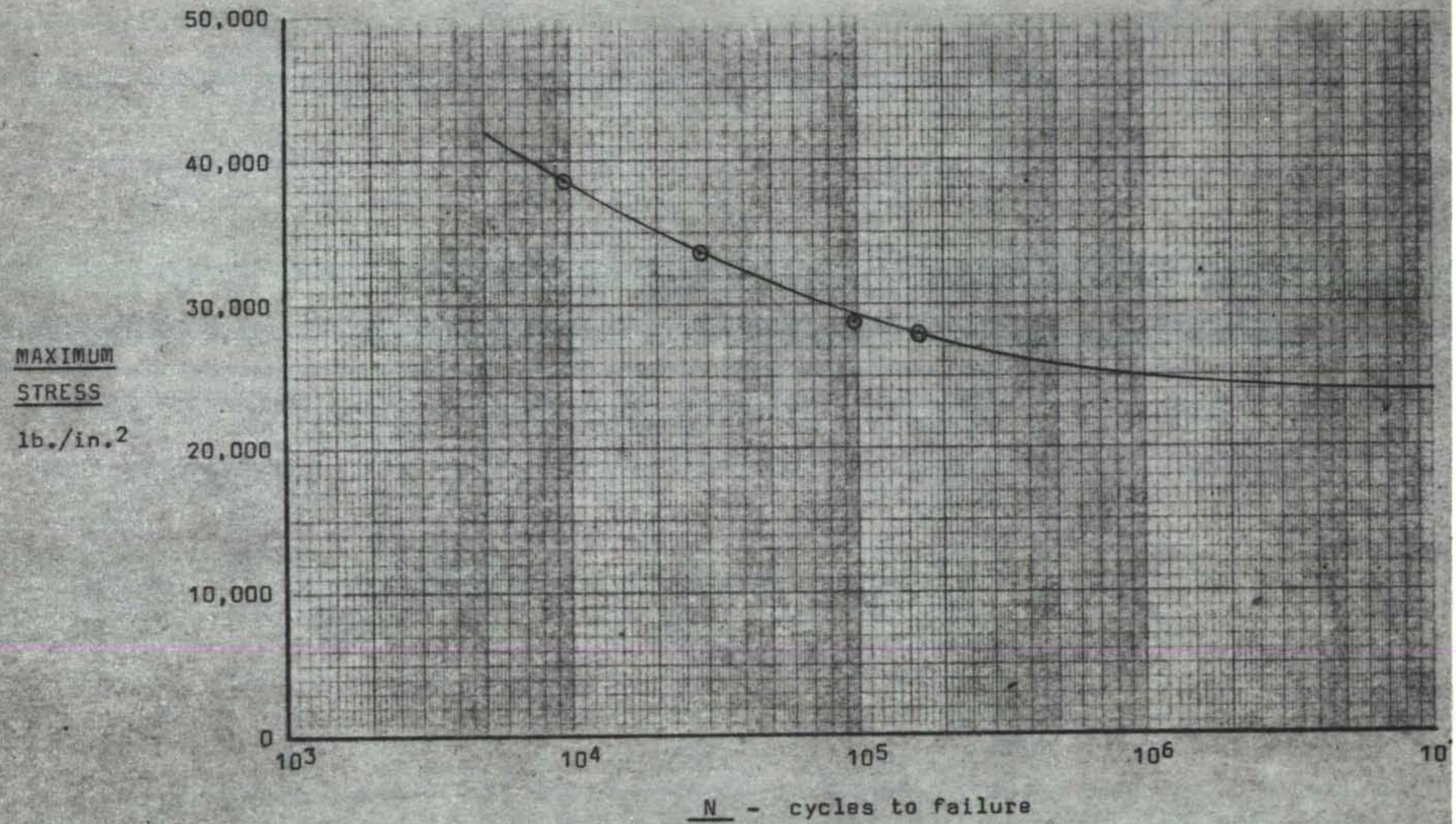

Figure 59. Test No. 11. Loaded Hole Control. 20,000 psi Mean Stress.

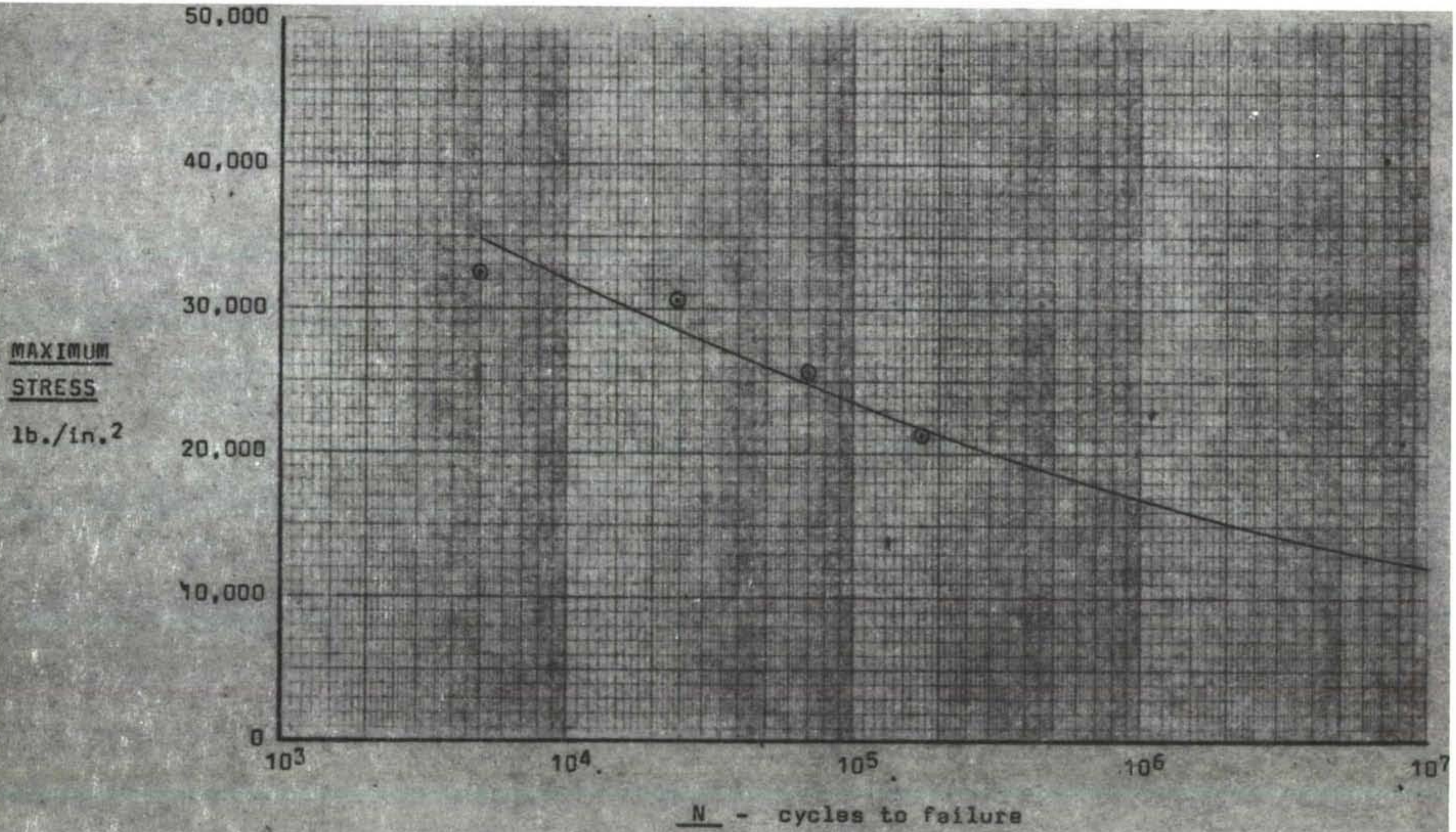

Figure 60. Test No. 10. Loaded Hole Control. 10,000 psi Mean Stress.

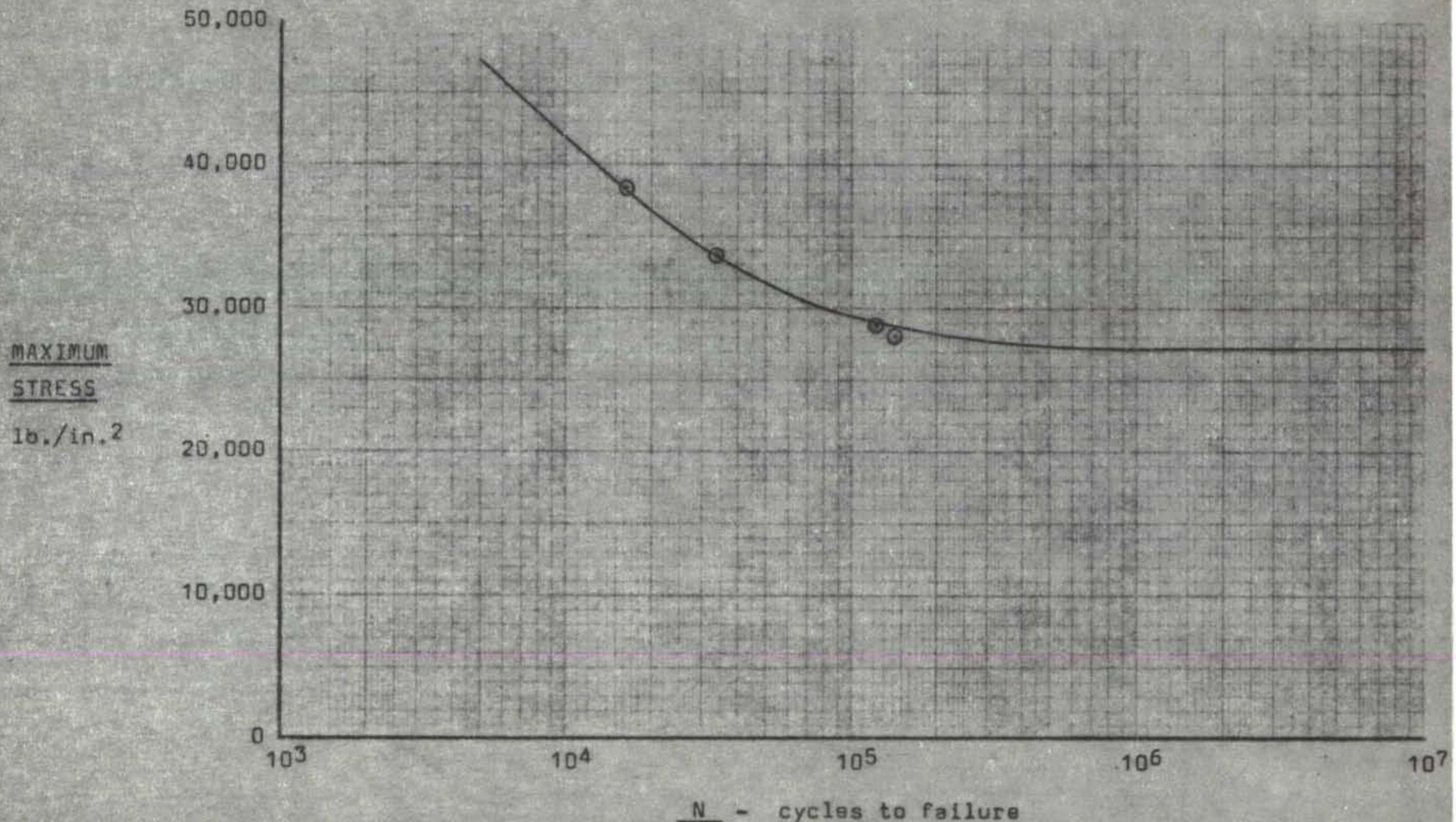

Figure 63. Test No. 22. Loaded Hole Reaming. Hole Diameter Increase, .03 in., Ream at 50% Life.

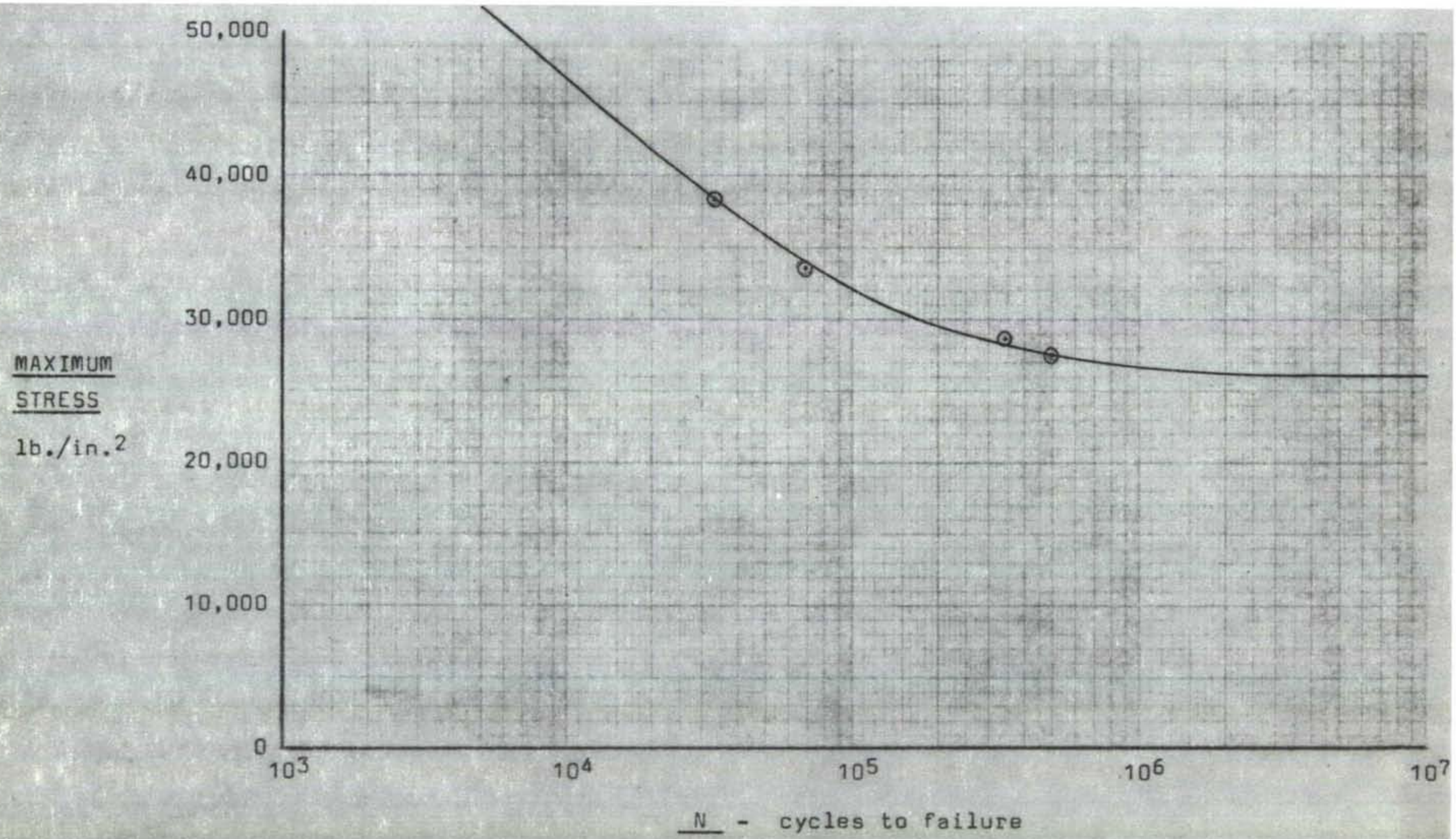

Figure 62.. Test No. 23. Loaded Hole Reaming. Hole Diameter Increase, .06 in., Ream at 50% Life.

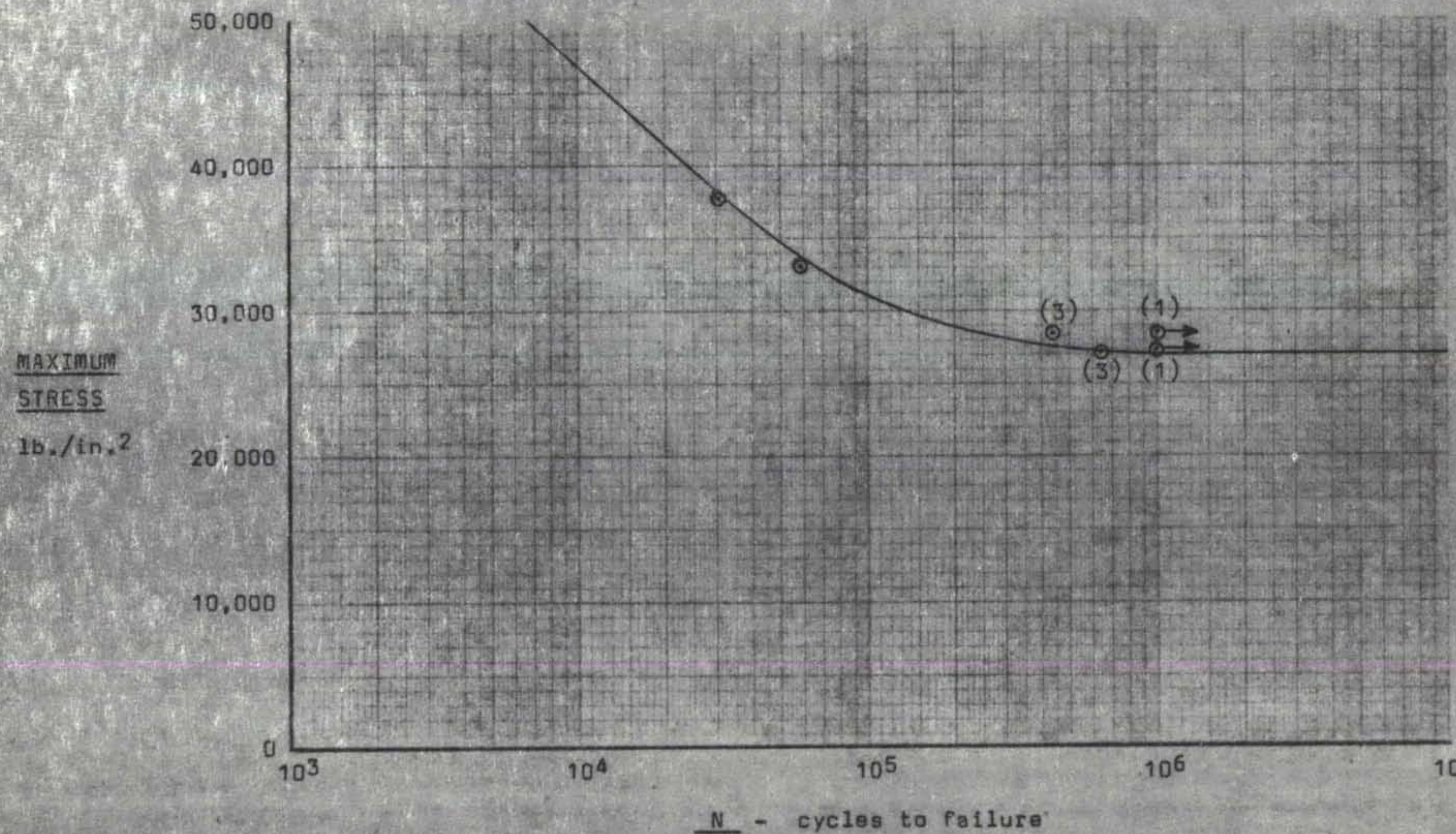

Figure 63. Test No. 27. Loaded Hole Peening (.007A).

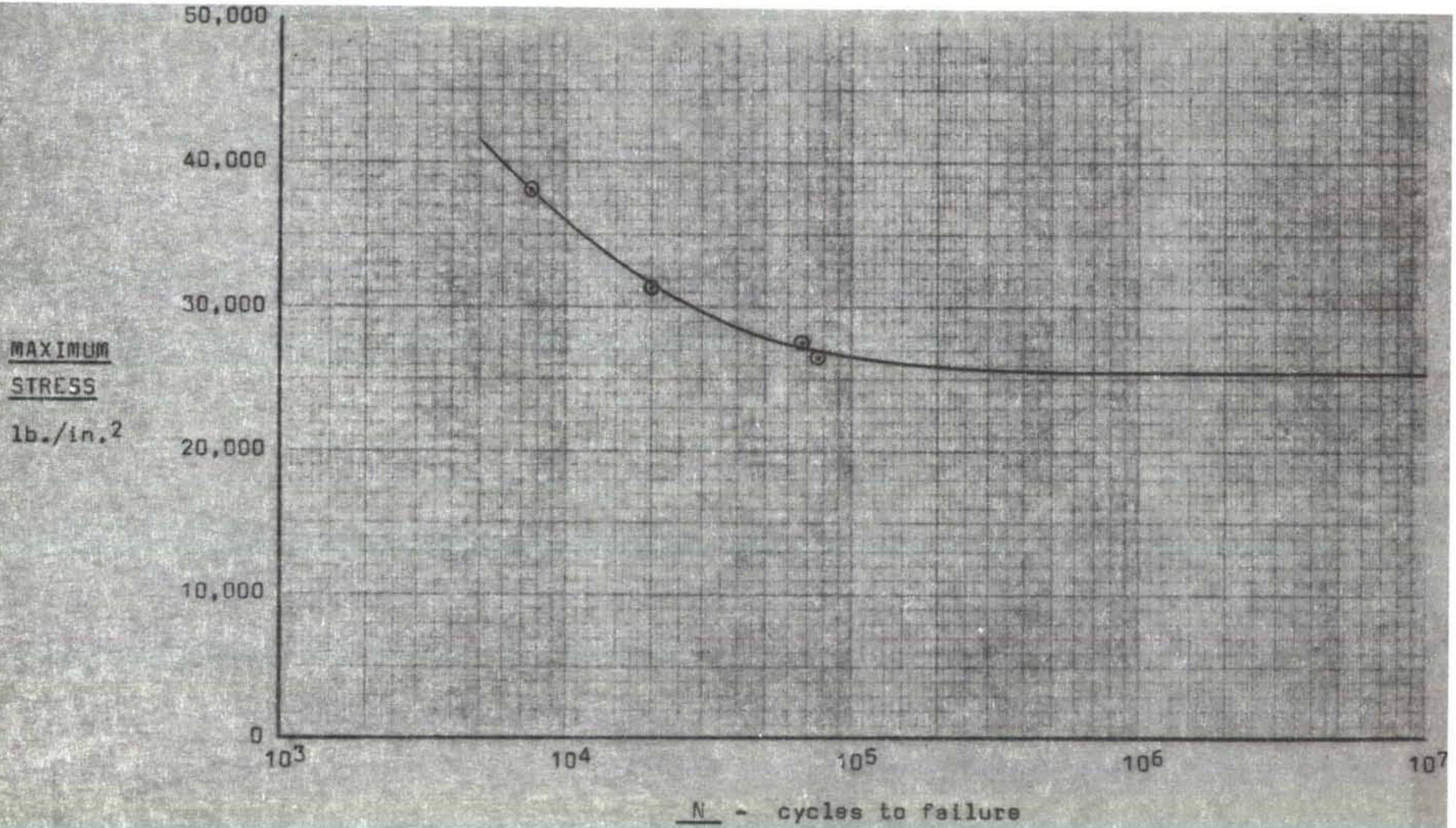

Figure 64. Test No. 13. 4 Bolt Joint Control. 20,000 psi Mean Stress.

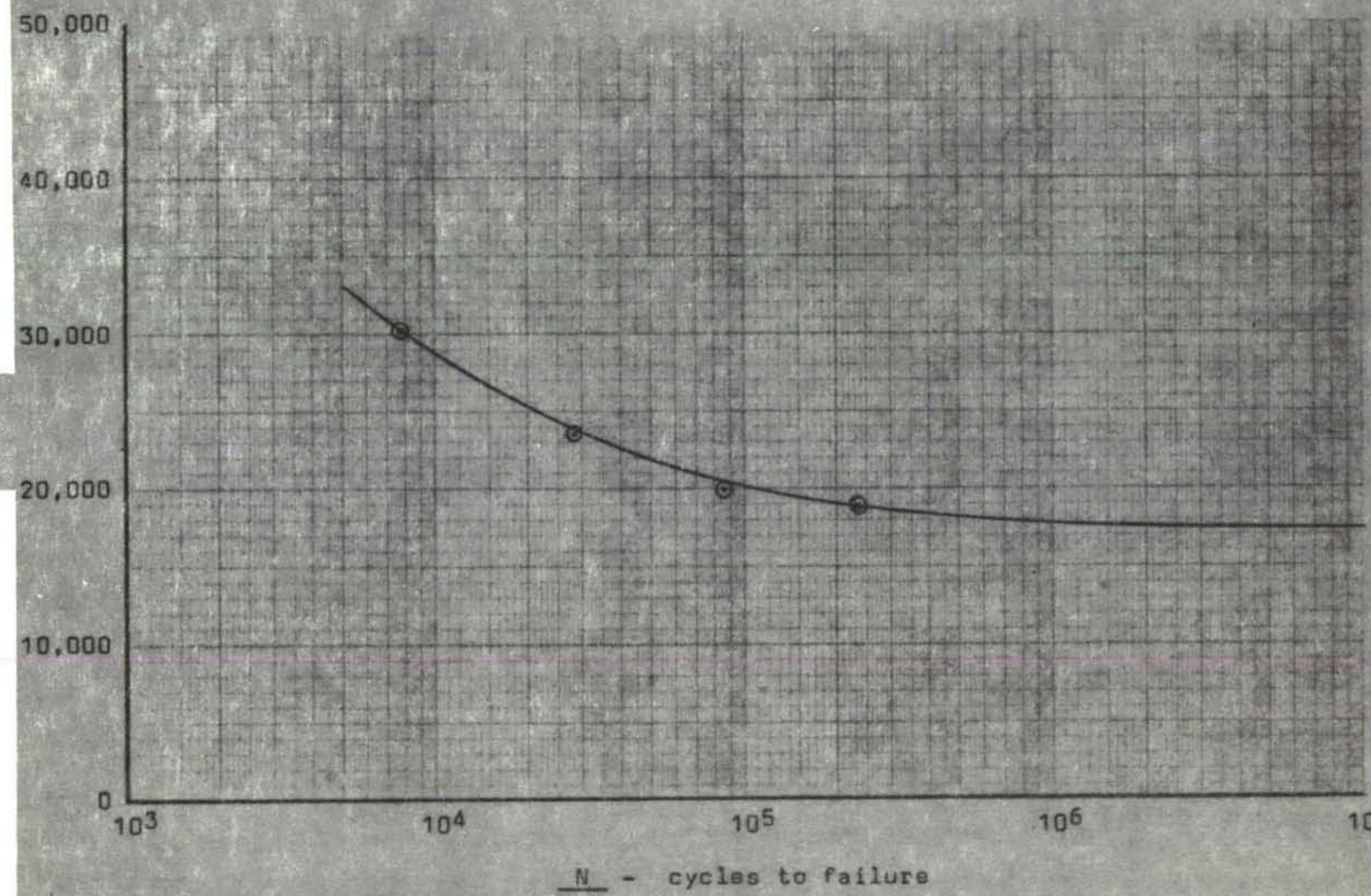

Figure B65. Test No. 12. 10,000 psi Mean Stress. 4 Bolt Joint Control.

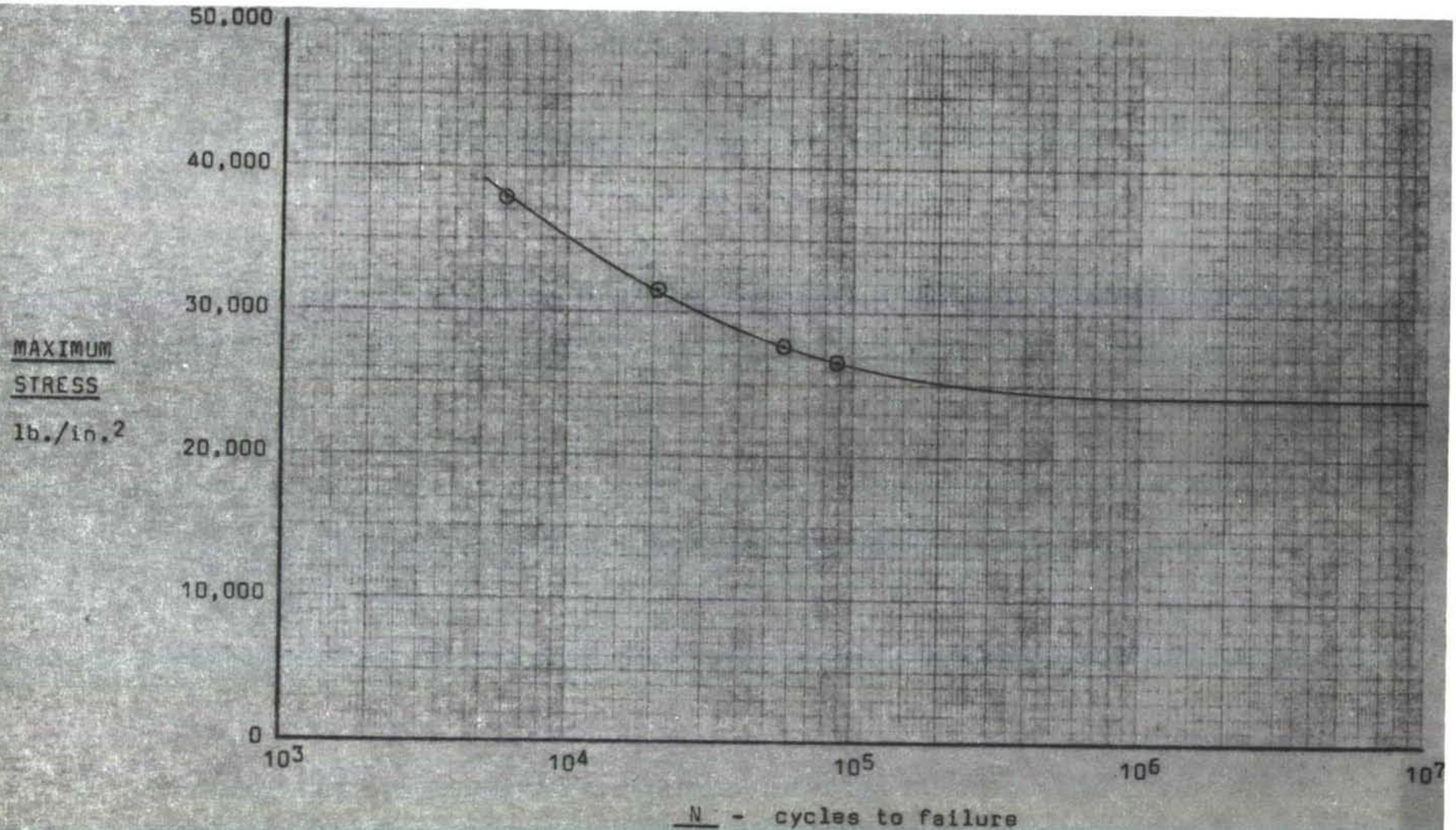

Figure 66.. Test No. 24. 4 Bolt Joint Reaming. Hole Diameter Increase, .03 in. Ream at 50% Life.

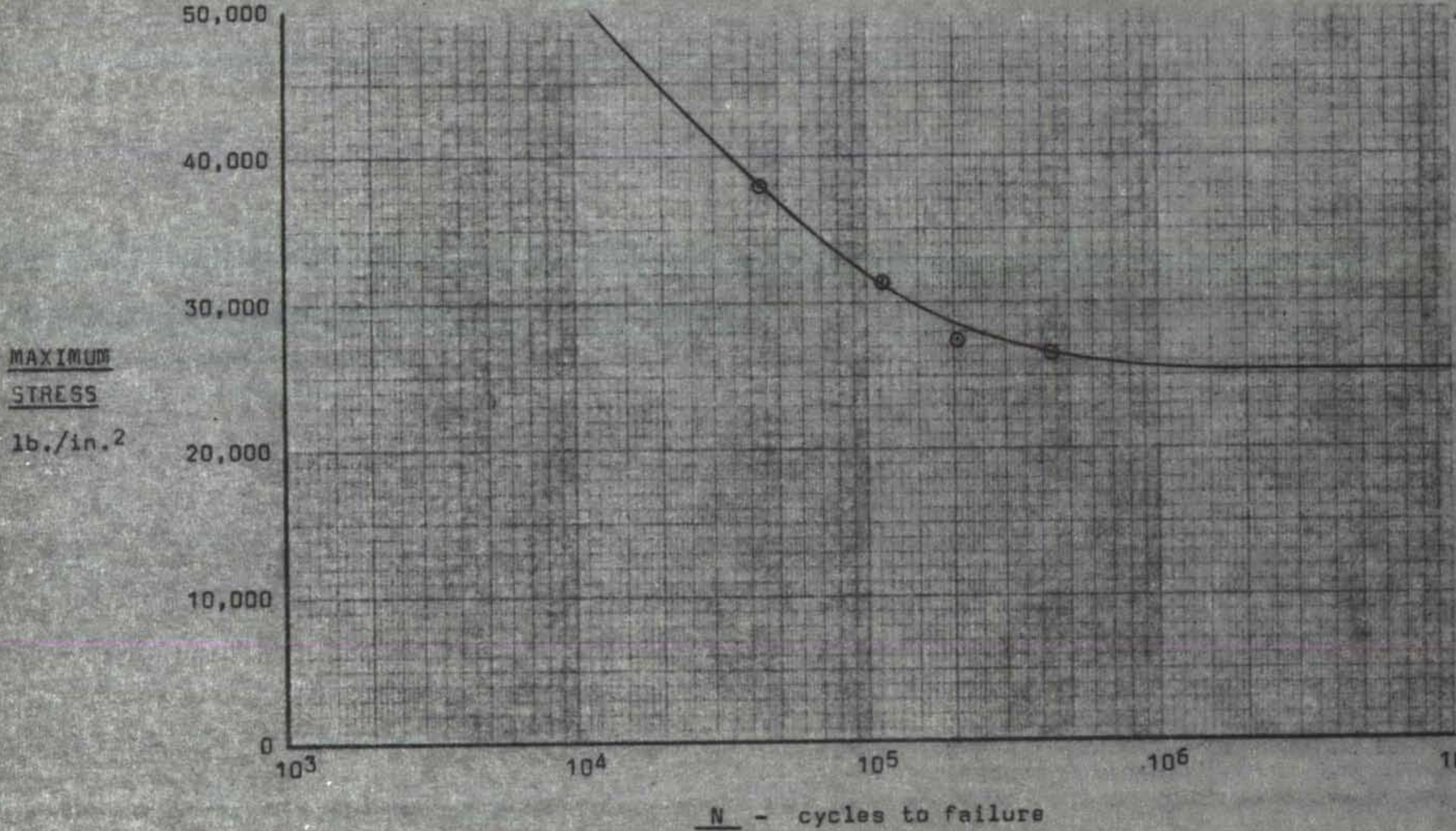

Figure 67. Test No. 25. 4 Bolt Joint Reaming. Hole Diameter Increase, .06 in. Ream at 50% Life.

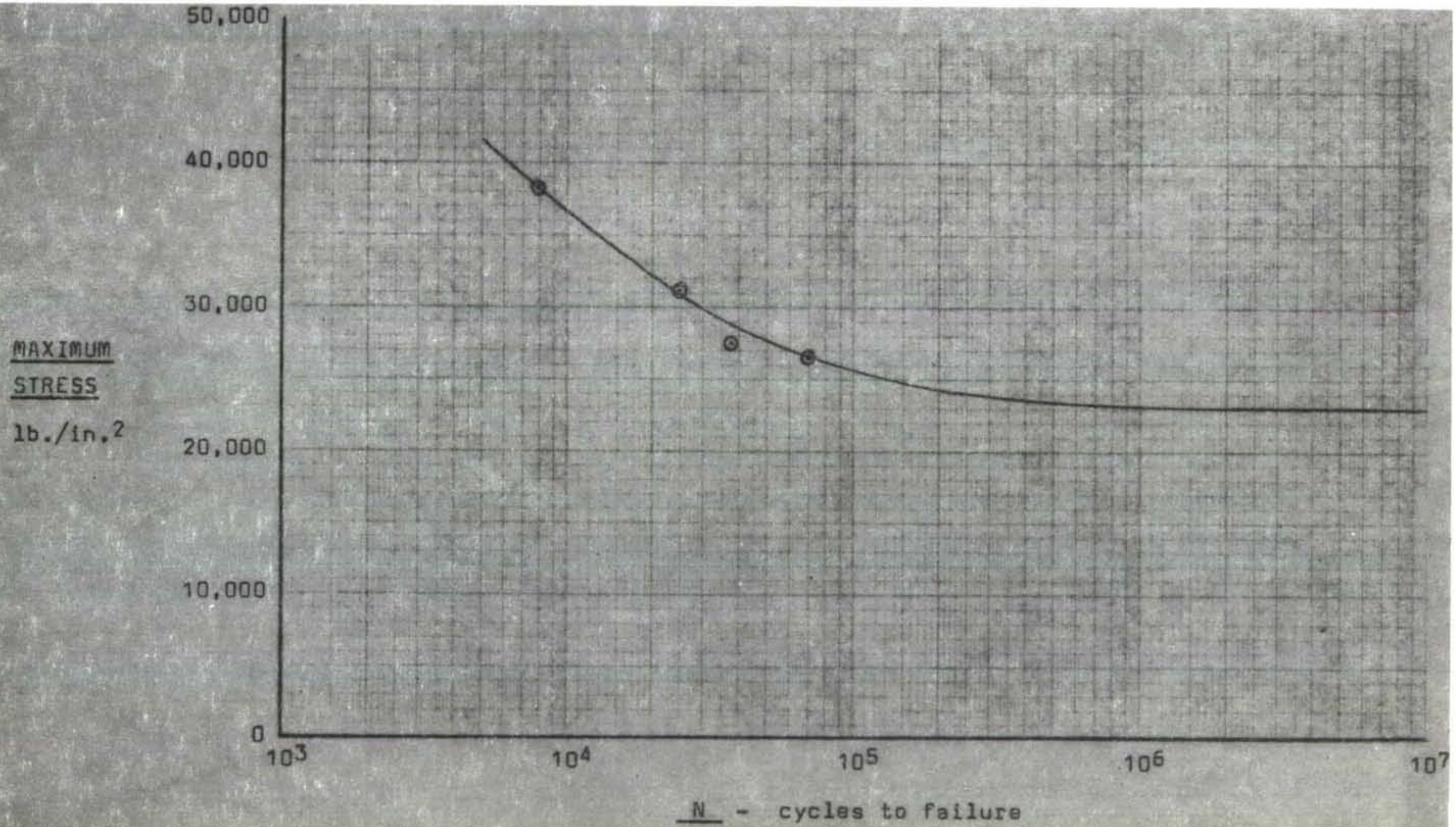

Figure 68. Test No. 31. 4 Bolt Joint Reaming, Low Bolt Preload. Hole Diameter Increase, .03 in. Bolt Torque, 12 inch-pounds. Ream at 50% Life.

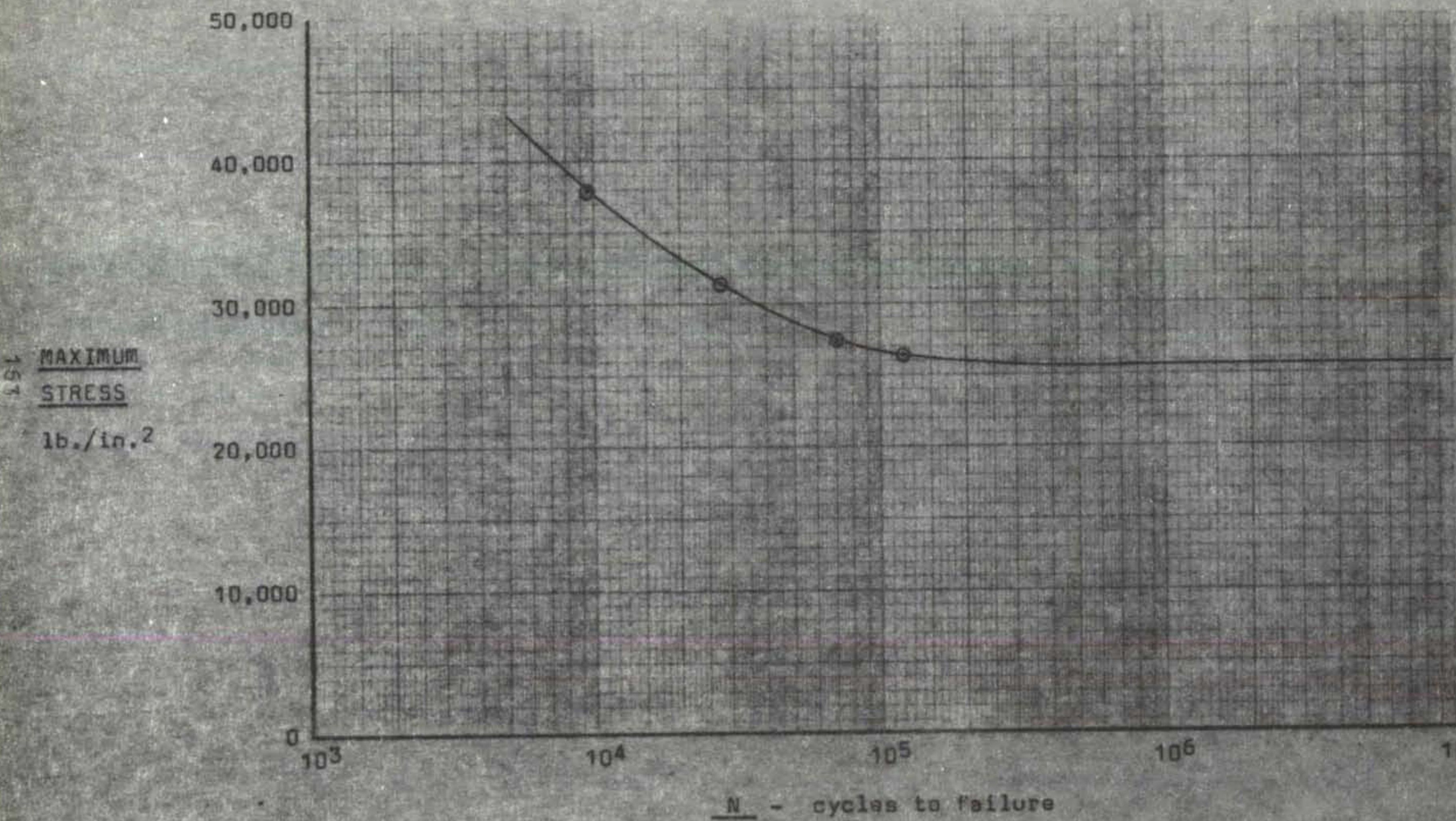

Figure 69. Test No. 32. 4 Bolt Joint Reaming, Low Bolt Preload. Hole Diameter Increase, .06 in. Bolt Torque 12 inch-pounds. Ream at 50% Life.

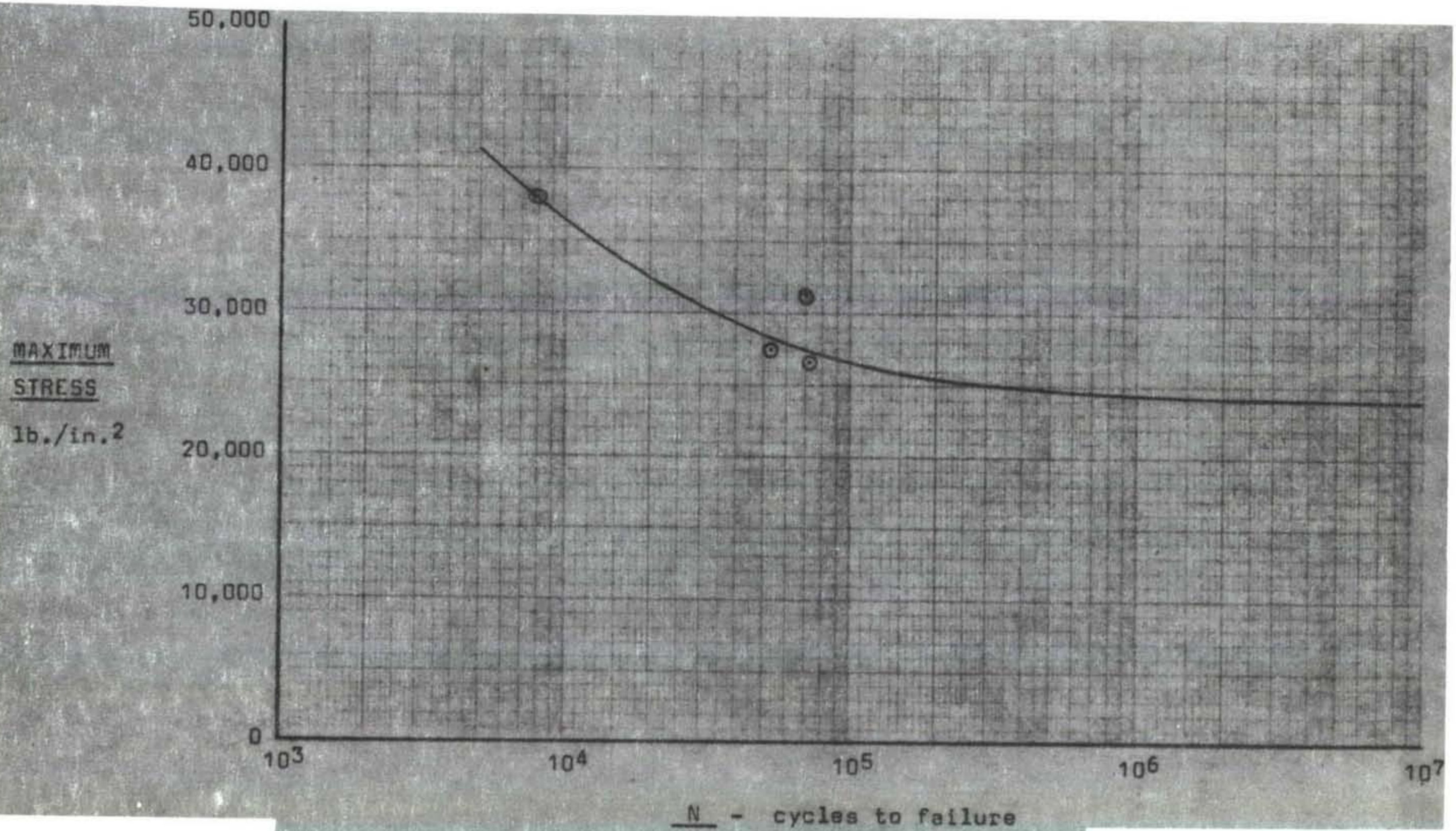

Figure 70. Test No. 28. 4 Bolt Joint Peening (.007A).

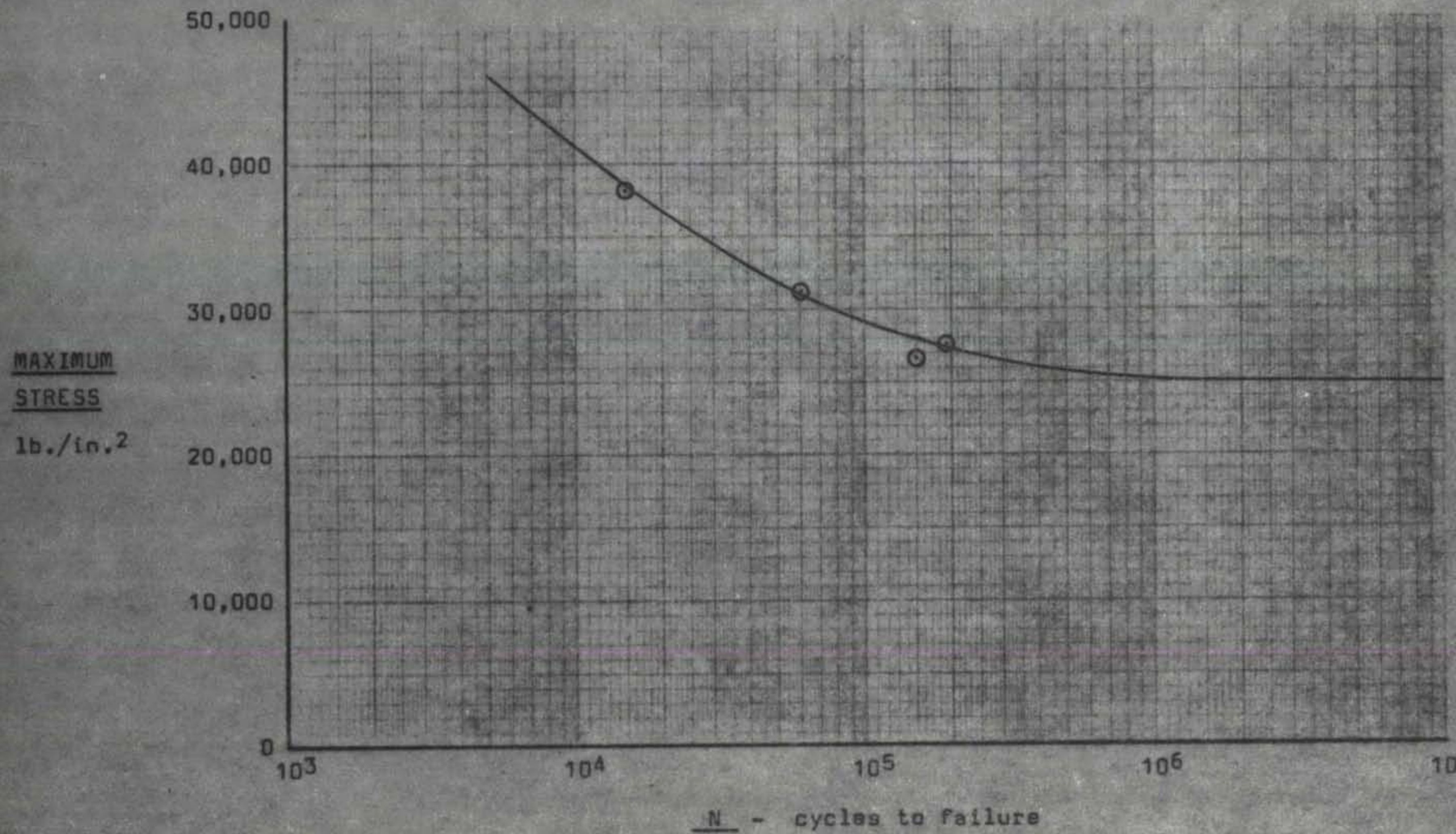

Figure B74. Test No. 46. 8 Bolt Joint Control. 20,000 psi Mean Stress.

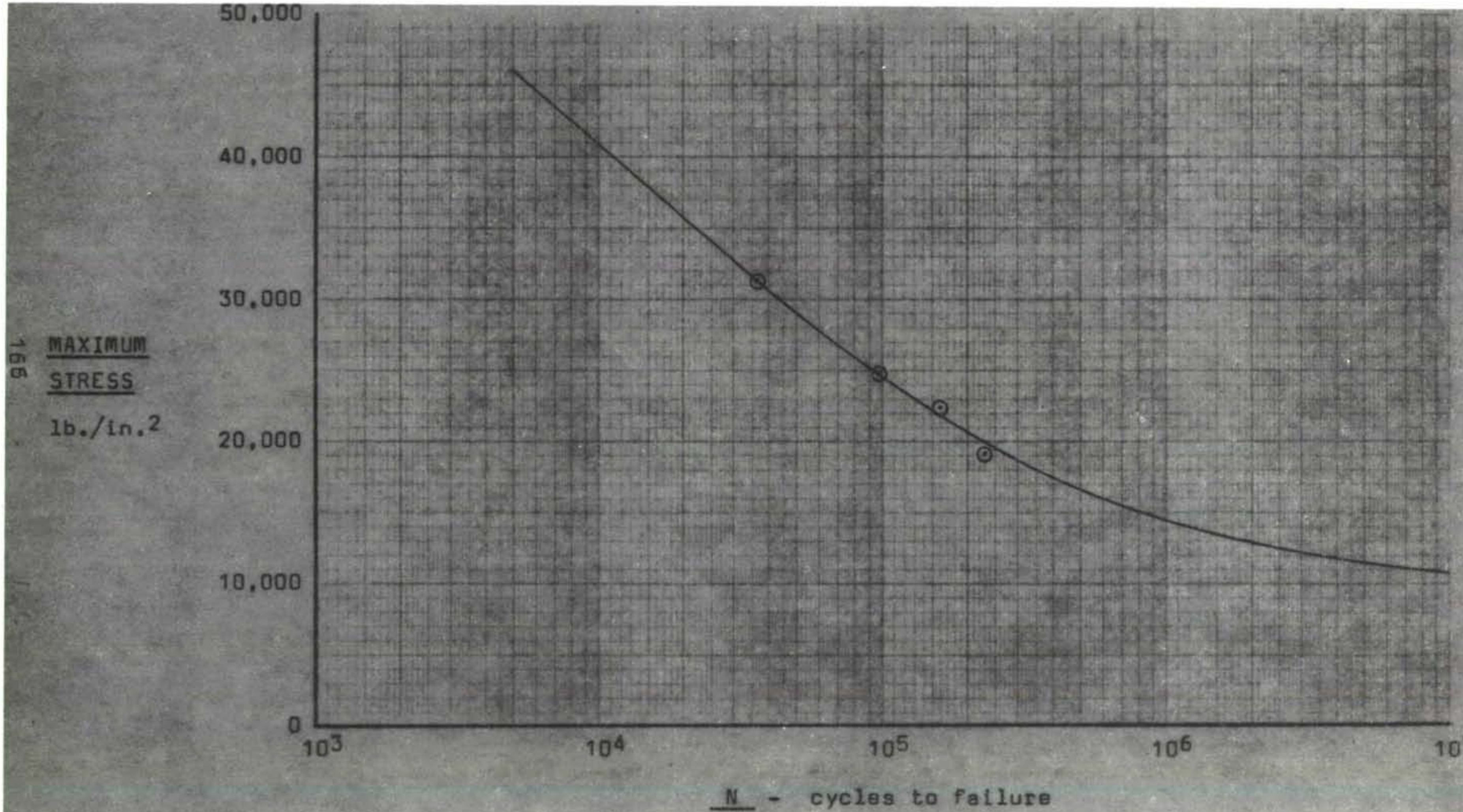

Figure B72. Test No. 48. 10,000 psi Mean Stress Control.

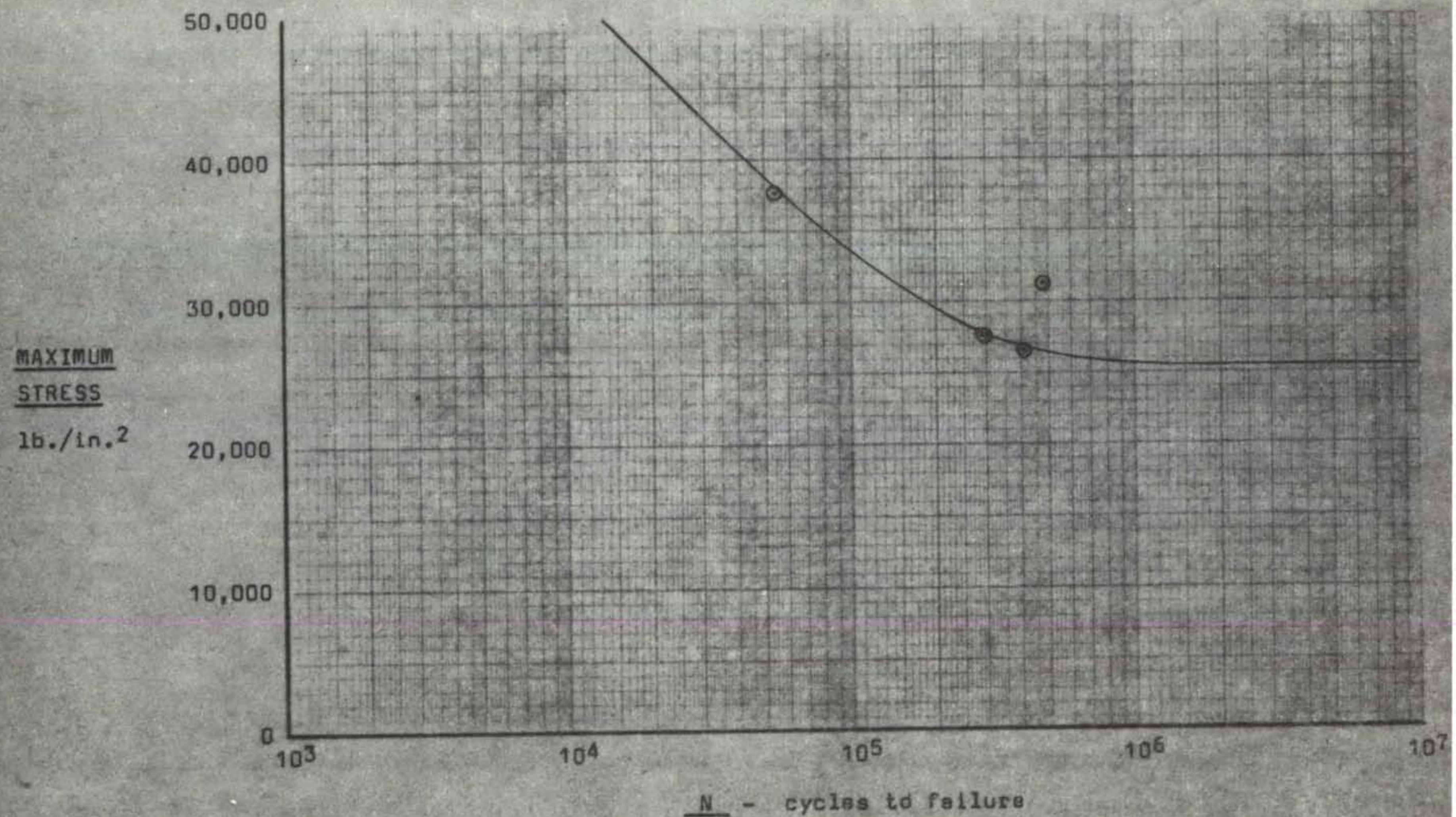

Figure 73. Test No. 47. Hole Diameter Increase, .06 in. Ream and Peen at 50% Life.

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11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY Air Force Flight Dynamics Laboratory (FDTR) Wright-Patterson AFB, Ohio 45433

13. ABSTRACT

This report presents the results and conclusions of a specimen testing program established to confirm or modify certain conclusions reached during the cyclic test of the A-26A wing and wich affect the A-26A Airplane Service Fatigue Life Prediction.

The object of the program was to evaluate the effects of reaming existing fatigue-critical bolt holes to larger diameters and peening the metal surfaces inside of and adjacent to the enlarged holes.

Specimens were designed to duplicate the conditions of the fatigue-critical portions of the A-26A wing. A series of tests were run, changes were made in the program schedule as the result of information gained, and a final series of tests were conducted.

It was concluded that (1) the damage reduction due to the reaming process produced results very nearly as originally considered in the A-26A Service Life Prediction, and (2) the reduction in damage accumulation rates of the A-26A fatigue test wing, originally attributed to the effects of peening, was actually caused by an increase in bolt preload achieved upon installing larger diameter bolts after the reaming process.

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14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
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Shot Peening						
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